NASA Contractor Report 4375

1989 High-Speed Civil Transport Studies

HSCT Concept Development Group
Advanced Commercial Programs

CONTRACT NAS1-18378
MAY 1991

NASA Contractor Report 4375

1989 High-Speed Civil Transport Studies

HSCT Concept Development Group
Advanced Commercial Programs
Douglas Aircraft Company
Long Beach, California

Prepared for
Langley Research Center
under Contract NAS1-18378

NASA
National Aeronautics and
Space Administration
Office of Management
Scientific and Technical
Information Division
1991

ABSTRACT

This report contains the results of the Douglas Aircraft Company system studies related to high-speed civil transports (HSCTs). The tasks were performed under a 1-year extension of NASA Langley Research Center Contract NAS1-18378.

The system studies were conducted to assess the environmental compatibility of a high-speed civil transport at a design Mach number of 3.2. Sonic boom minimization, exterior noise, and engine emissions have been assessed together with the effect of laminar flow control (LFC) technology on vehicle gross weight.

The general results indicated that (1) achievement of a 90-PLdB sonic boom loudness level goal at Mach 3.2 may not be practicable, (2) the high-flow engine cycle concept shows promise of achieving the sideline FAR Part 36 noise limit but may not achieve the aircraft range design goal of 6,500 nautical miles, (3) the rich-burn/quick-quench (RB/QQ) combustor concept shows promise for achieving low $EINO_X$ levels when combined with a premixed pilot stage/advanced-technology high-power stage duct burner in the P&W variable-stream-control engine (VSCE), and (4) full-chord wing LFC has significant performance and economic advantages relative to the turbulent wing baseline.

FOREWORD

The High-Speed Civil Transport Study Phase IIIA was a 1-year extension of the previous 2 years' work (Phases I to III). Phase IIIA was a combined technical research activity and systems evaluation covering the period from 1 October 1988 to 30 September 1989.

Work was accomplished as a task order activity by Douglas Aircraft Company in Long Beach, California. This work was under the direction of the NASA Langley Research Center, Hampton, Virginia, and was jointly funded under Contract NAS1–18378.

The NASA Contracting Officer Technical Representative was Charles E. K. Morris, Jr., for the early part of the study and Donald L. Maiden for the latter part. The Douglas program manager was Donald A. Graf, Business Unit Manager, HSCT Concepts. Principal investigators were H. Robert Welge, technology integration and product definition and assistant program manager; Gordon L. Hamilton, concept assessment; Bruce W. Kimoto, requirements integration; and Alan K. Mortlock, environmental assessment. Expert assistance was provided by Richard T. Cathers, configuration integration; Cliff Y. Kam, structures and materials; John W. Stroup, market research; Marc L. Schoen, airports; and Maurice Platte, manufacturing and development costing. McDonnell Aircraft Company technical staff provided consultation relating to NASP technologies.

Other Douglas HSCT concept team members were:

Administration	E. C. Anderson, J. A. Harkins
Aerodynamics	G. A. Intemann, J. A. Page, Dr. D. L. Antani, Dr. A. G. Powell, P. Miller, J. Morgenstern, A. Killeen, D. Schowe, R. Sohn
Business Operations	M. L. Shell
Laminar Flow	W. E. Pearce, N. M. Jerstad
Product Support	R. E. Swartzbacker
Propulsion	F. R. Mastroly, J. A. Meyer
Structures and Materials	Dr. E. G. Chow, G. Vinluan
Weights	G. J. Espil

GLOSSARY

A	Area
A_E	Equivalent Area
Al MMC	Aluminum Metal Matrix Composite
AR	Aspect Ratio
BPR	Bypass Ratio
c/lb	Cents per Pound
C_D	Drag Coefficient
C_L	Lift Coefficient
$(C_L)_{max}$	Maximum Lift Coefficient
C_p	Pressure Coefficient
C_q	Suction Coefficient (ratio of mass flux through the skin to the freestream mass flux)
CASES	Computer-Aided Sizing and Evaluation System
CET	Combustor Exit Temperature
CFD	Computational Fluid Dynamics
CG	Center of Gravity
CR	Contractor Report
dB	Decibel (Reference Pressure Re. 20 μPa)
DAC	Douglas Aircraft Company
D/B	Duct Burner
DEG	Degrees
EAS	Equivalent Airspeed
$EINO_x$	NO_x Emissions Index (lb NO_x per 1,000 lb fuel burned)
EJ	Ejector
EPNdB	Unit of Effective Perceived Noise Level
ETA	Semispan Fraction Y/(b/2)
fps	Feet per second
ft	Feet
ft^2	Square Foot
ft^3	Cubic Foot
F	Fahrenheit
F_n	Thrust per Engine
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations
FPR	Fan Pressure Ratio
F-function	Whitham F-function used in sonic boom analysis
GAG	Ground-Air-Ground

GE	General Electric Aircraft Engines
h	Height
HP	Horsepower
HSCT	High-Speed Civil Transport
in	Inches
IATA	International Air Transport Association
ICAO	International Civic Aviation Organization
ISA	International Standard Atmosphere
IVP	Inverted Velocity Profile
lb	Pounds
L_{ce}	C-Weighted Sound Exposure Level in Decibels
L/D	Lift-to-Drag Ratio
$(L/D)_{max}$	Maximum Lift-to-Drag Ratio
LE	Leading Edge
LFC	Laminar Flow Control
M	Mach Number
M/E	Mixer Ejector
MLW	Maximum Landing Weight
MMC	Metal Matrix Composite
MSC/NASTRAN	MacNeal-Schwendler Corporation's Finite Elements Computer Program
MSEC	Millisecond
N MI	Nautical Miles
NASA	National Aeronautical and Space Administration
NO_x	Oxides of Nitrogen (all species)
N-Wave	Basic sonic boom waveform, so called because it resembles an "N"
OEW	Operator's Empty Weight
OPR	Overall Pressure Ratio
psf	Pound per Square Foot
P	Perceived Loudness
P&W	Pratt & Whitney
PLdB	Stevens Mark VII Perceived Level of Loudness in Decibels
PM/PV	Premixed/Prevaporized
Psh	Front Shock Overpressure
Q	Dynamic Pressure
R	Rankine
RB/QQ	Rich Burn/Quick Quench
RSR-Al	Rapid Solidification Rate Aluminum
sfc	Specific Fuel Consumption

S_{ref} and S_w	Wing Reference Area
S	Cell Size
SCS-8	Silicon Carbide Fibers
SEEB	Sonic Boom Minimization Computer Code, NASA LaRC
SL	Sea Level
SLS	Sea Level Static
SLST	Sea Level Static Thrust
Supp	Suppressor
t_r	Core Ribbon Thickness
T	Temperature
T_3	Compressor Discharge Temperature
T_4	Turbine Entry Temperature
TAD	Technology Availability Date
TAS	True Airspeed
TBE	Turbine Bypass Engine
TOFL	Takeoff Field Length
TOGW	Takeoff Gross Weight
TSFC	Thrust Specific Fuel Consumption
V_J	Exhaust Jet Velocity
VCE	Variable Cycle Engine
VSCE	Variable-Stream-Control Engine
W	Weight
X/C	X-Coordinate/Chord
Y/C	Y-Coordinate/Chord
2-D	Two Dimensional
3-D	Three Dimensional
δ_f	Wing Trailing Edge Flap Deflection Angle
Δp	Overpressure
ϕ_i	Equivalent Ratio (local fuel per air ratio to stoichiometric ratio)
ρ_c	Core Density
t/c	Thickness-to-Chord ratio
t/c_{max}	Maximum Thickness-to-Chord Ratio

CONTENTS

CONTENTS (CONTINUED)

FIGURES

TABLES

SECTION 1
SUMMARY

This report contains results of a Douglas Aircraft Company study to assess the environmental compatibility (e.g., sonic boom, exterior noise, and engine emissions) of a high-speed civil transport (HSCT). Integration of laminar flow control technology was also investigated. These studies were conducted at a design Mach number of 3.2. The baseline vehicle concept, which is described in the appendix, carries 300 passengers with a range objective of 6,500 nautical miles. Pertinent results from the previous 2 years of study are included in the next section.

The environmental status, general comments, and recommendation for further studies are as follows:

Sonic Boom Minimization — A tentative sonic boom loudness acceptability goal of 90 PLdB was selected based on human response data analyzed by Wyle Laboratories under contract to Douglas. Vehicle design studies progressed to the point where the sonic boom loudness of the Mach 3.2 aircraft was reduced from 102 PLdB to 96.5 PLdB by incorporating a canard in the design. The actual aircraft cruise weight for Mach 3.2 may have to be restricted to achieve the 90-PLdB goal. Douglas recommends that (1) a sonic boom sensitivity study be conducted to determine the effects of Mach number on achieving 90 PLdB, (2) an appropriate Mach number be selected and sonic boom minimization designs continued, and (3) human response studies be conducted to verify PLdB as the appropriate noise metric and 90 PLdB as an acceptable level.

Exterior Noise — The exterior noise goal was selected to meet the current subsonic FAR Part 36/ICAO Annex 16 Chapter 3 noise certification limits. This goal was selected to establish the acoustic technology necessary to be compatible with long-range subsonic aircraft now in production. Achievement of this goal should produce a minimal impact on airport community noise. Various engine cycles and noise suppression systems have been evaluated from both General Electric (GE) and Pratt & Whitney (P&W). Some engine configurations show promise in achieving the FAR Part 36 sideline noise requirement, which appears to be the most difficult to achieve. The sideline noise requirement limits the design range goal to approximately 5,500 nautical miles with currently proposed "noise" technology. Certain engine cycles achieve the design range goal of 6,500 nautical miles but are 5-6 EPNdB above the noise requirement. It is recommended that studies of engine cycle, noise suppression, and vehicle high-lift geometry be continued to achieve the combined 6,500-nautical-mile range goal and Stage 3 noise requirement. Operational procedures to minimize airport noise should be considered as an integral part of these studies.

Engine Emissions — To date, engine emissions design criteria have not been established. Therefore, an interim low NO_x combustor design goal of 5-10 $EINO_x$ has been proposed to facilitate timely research and development validation by the engine companies. The rich-burn/quick-quench (RB/QQ) combustor concept shows promise for achieving low $EINO_x$ levels (e.g., 5-10 range) when combined with a premixed pilot-stage/advanced-technology high-power stage duct burner in the Pratt & Whitney variable-stream-control engine (VSCE). It is recommended that the engine companies continue research to develop innovative low-NO_x combustors and, correspondingly, that atmospheric modelers continue HSCT fleet studies of exhaust emissions effects on ozone. Further, the airframers should continue vehicle configuration trade studies and mission profile optimization studies.

Laminar Flow Control (LFC) — Systems integration studies of LFC technology showed that full-chord wing LFC would have performance and economic advantages relative to the partial-chord wing LFC concept and turbulent baseline. More in-depth definitions of the LFC suction compressors, ducting, and integration with the airframe are recommended. Trade studies of structural concepts integrated with LFC should be conducted to identify the most practical approaches.

General Recommendations — Further definition of structural materials appropriate for HSCT technology goals (e.g., strength, weight, durability, producibility) is recommended. Further reductions in the vehicle weight are of prime importance for reduced takeoff gross weight and fuel burn as well as minimization of sonic boom and lower airport noise levels.

An international plan needs to be developed with the appropriate government agencies to define noise and emission standards for the HSCT. Public awareness on progress toward achieving an environmentally acceptable aircraft is considered to be an important element in the successful development of the HSCT.

In general, the Mach 3.2 configuration looks promising regarding aircraft performance. Minimizing sonic boom to the 90-PLdB level is difficult and will require alternative approaches beyond configuration shaping at the cruise Mach number. Further studies are recommended to identify additional technology required to achieve a "realistic" configuration that is environmentally acceptable and economically viable.

SECTION 2
INTRODUCTION

This report presents the results of studies conducted as part of a continuing Douglas and NASA effort to determine the technologies required for the next-generation supersonic transport. This work (Phase IIIA) represents an extension of the 2 previous years' activities (Phases I through III) covering technical, environmental, and economic aspects of the HSCT. These previous phases led to focused studies to find a solution to the environmental issues of sonic boom, exterior noise, and engine emissions. Laminar flow control (LFC) technology was studied to determine the impact on vehicle gross weight reduction (fuel burn reduction), incuding installation requirements such as ducting, pumps, etc.

During earlier phases conducted by Douglas under NASA contract, conceptual vehicle definitions were developed over a range of Mach numbers from 2 to 25 (employing fuels from Jet A to hydrogen). In particular, the commercial value, mission performance, environmental compliance, and technology requirements were evaluated. This led to the following conclusions:

- Market projections for the 2000 to 2025 time period indicate sufficient passenger traffic for ranges beyond 2,000 nautical miles to support a fleet of economically viable and environmentally compatible high-speed commercial transports. Fleet needs, considering a 300-seat aircraft, could total 1,500 or more by 2025.
- The Pacific Rim area will become the major traffic region after the year 2000, leading to a design range objective of 6,500 nautical miles.
- Economic viability places emphasis on environmentally acceptable overland supersonic flight. The constraint of no overland supersonic flight reduces potential aircraft productivity and thus increases aircraft operating costs.
- Aircraft productivity increases with cruise speed up to about Mach 5 to Mach 6 for market applications ranging from 2,000 to 6,500 nautical miles. Above this point, the relative significance of cruise speed diminishes, and productivity is virtually constant.

SECTION 3
SONIC BOOM

3.1 INTRODUCTION

The sonic boom studies conducted at Douglas in the previous phases of HSCT research were primarily concerned with assessing sonic boom levels for the Mach 3.2 and Mach 5.0 aircraft configurations and establishing preliminary sonic boom acceptability criteria. Inasmuch as these early studies formed the basis for this study, it is appropriate to briefly review the important conclusions.

The passenger and routing forecasts for the time period during which the HSCT would enter service will control the economics of the aircraft. Hence, the critical design parameters will be speed and range. Based on work to date, the baseline for this study is a Mach 3.2, 6,500-nautical-mile aircraft designed to carry 300 passengers. An important design parameter for sonic boom is aircraft length. The length of the aircraft is primarily constrained by the longitudinal bending characteristics of the fuselage and airport compatibility concerns. Previous studies indicated that a length of approximately 300 feet is appropriate for this class of vehicle.

Three different noise metrics were identified for assessing the relative acceptability of sonic booms on the ground: front shock overpressure (p_{sh}), perceived loudness (P), and C-weighted sound exposure level (L_{ce}). The limited acceptability research conducted to date has not identified any one single metric as the best overall. Therefore, to avoid a premature selection, three metrics were maintained to cover all aspects of boom annoyance. After a comprehensive survey of the existing data, conservative criteria were established with these metrics for the purpose of setting design goals for future sonic boom minimization studies. The design goals are as follows: 90-PLdB perceived loudness, 102-dB C-weighted sound exposure level. These levels are approximately equal to a 0.6-psf front shock overpressure.

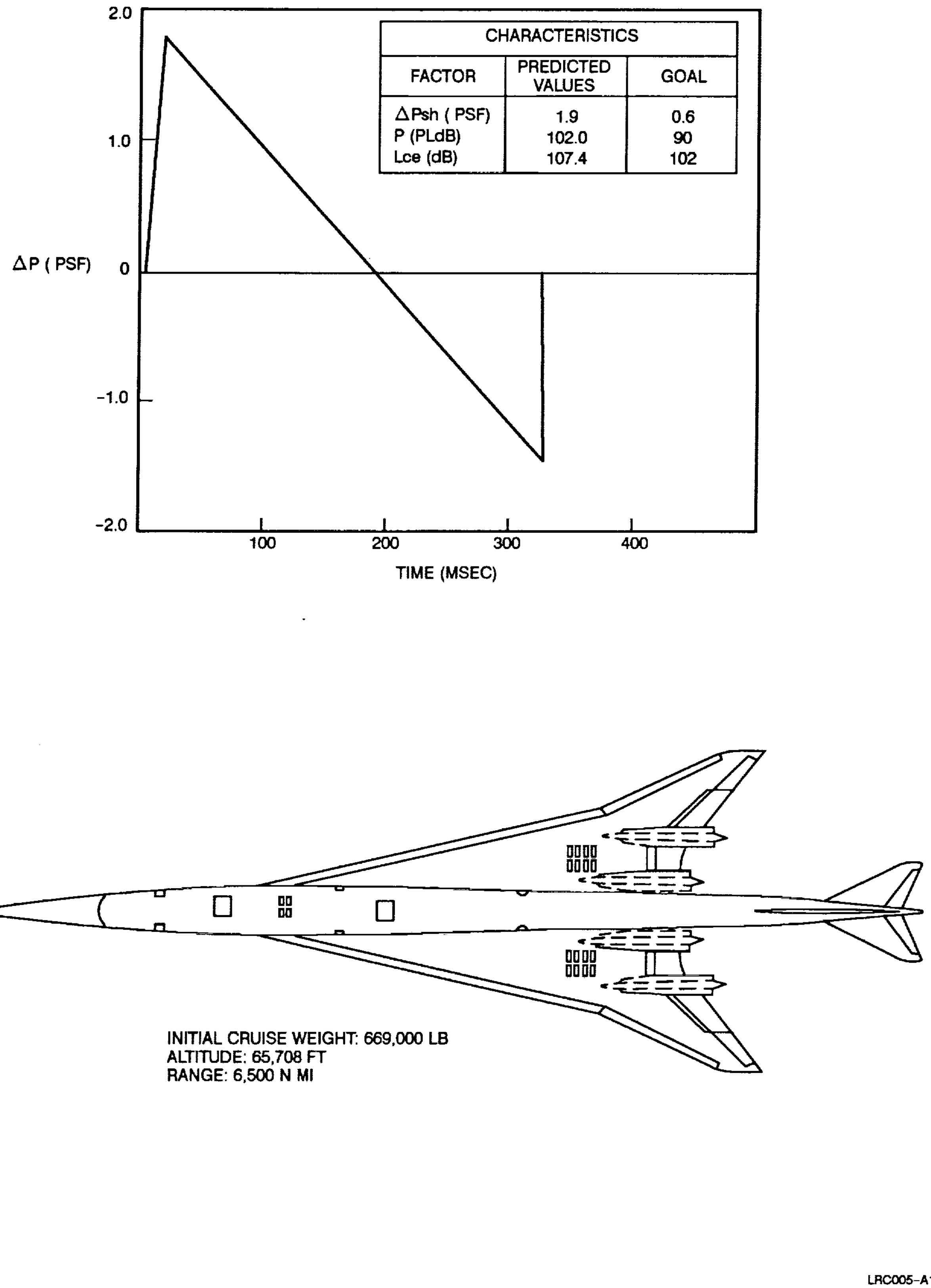
During the early phases of HSCT studies, sonic boom levels were assessed but were not a part of the design process. Aircraft characteristics such as weight, range, and lift-to-drag ratio were considered in an attempt to optimize the overall mission performance. Not surprisingly, the unconstrained sonic boom levels were typically well above the design goals. Figure 3-1 compares the Mach 3.2-3A (D3.2-3A) baseline sonic boom levels with the goals. This configuration (described in the appendix) served as the departure point for the minimization studies discussed in this section and presented the formidable design challenge of achieving a 12-PLdB reduction in loudness of the waveform on the ground.

Whereas the early sonic boom analysis was concerned with assessing a fixed configuration, this work examines a wide variety of aircraft shapes with the goal of minimizing the sonic boom level to 90 PLdB. The study includes both generic minimization concepts and the integration of these concepts into viable aircraft configurations.

3.2 APPROACH

There are two different approaches to minimizing sonic boom. The distinction between the two approaches is important because the optimal aircraft shape and operating conditions for each method are different.

The first and most straightforward approach is to minimize the initial pressure disturbance at the aircraft source. This is accomplished by making the aircraft both lightweight and slender. No attempt is made to shape the pressure field. As a result, the waveform on the ground is usually a basic N-wave



CHARACTERISTICS		
FACTOR	PREDICTED VALUES	GOAL
ΔPsh (PSF)	1.9	0.6
P (PLdB)	102.0	90
Lce (dB)	107.4	102

FIGURE 3-1. D3.2-3A BASELINE CONFIGURATION

or a close approximation. For this reason, the source disturbance minimization approach is often referred to as N-wave minimization. Most, if not all, of the early sonic boom minimization attempts used this approach.

The second approach developed as researchers became more familiar with sonic booms and the mechanisms by which booms annoy or startle people. It became apparent that parameters other than peak overpressure of the boom influence subjective human response. Seebass and George (Reference 3-1) realized that it is possible to take advantage of this fact and shape the aircraft planform so as to generate so-called "minimized" waveforms as depicted in Figure 3-2. These waveforms reduce the strength of the shock systems for a fixed amount of energy in the waveform. This has the effect of shifting the frequency distribution of the boom energy downward and reducing its perceived loudness. Seebass and George were also able to determine the F-function required to produce a minimized waveform for a set of given operating conditions (Reference 3-1). Since the F-function is directly related to the aircraft geometry, it is possible to specify an equivalent area distribution that will theoretically yield a minimized waveform on the ground. Darden has formalized this process into the SEEB computer program, which can be used as a design tool in sonic boom studies (Reference 3-2).

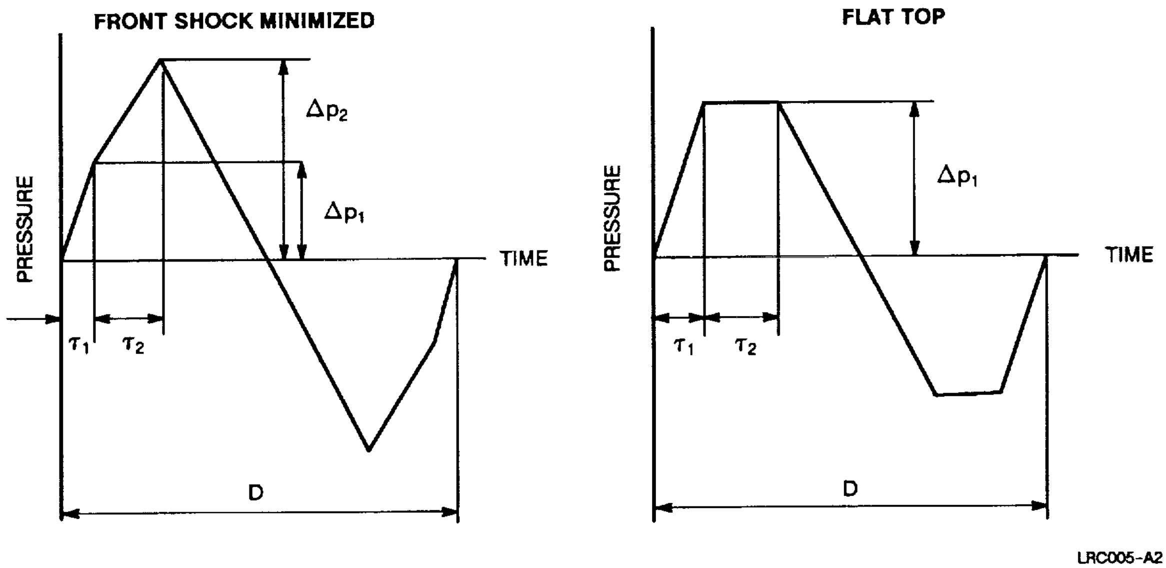


FIGURE 3-2. MINIMIZED SONIC BOOM WAVEFORMS

This second approach to sonic boom minimization is called waveform shaping. The principles that apply to boom shaping are well understood, but these principles have not yet been successfully synthesized into a viable aircraft configuration that meets both aerodynamic performance and economic criteria.

This study pursued minimization through both approaches. Two different configurations were developed, one for each approach, and compared for both sonic boom level and economic performance. This study assigned the highest design priority to the sonic boom characteristics of the vehicle, as opposed to generating aerodynamically optimized, sonic boom unconstrained vehicles as in the past study.

3.3 SOURCE DISTURBANCE MINIMIZATION

The principles of source disturbance minimization are straightforward and fall in line with classical supersonic design concepts. Highly swept wings and long, slender bodies are used to minimize the

drag and avoid large pressure disturbances. This is consistent with typical design practices, and an aircraft that has been optimized aerodynamically for supersonic cruise is already on its way to achieving these goals.

3.3.1 D3.2-5 Configuration

A new configuration, the D3.2-5, was developed as a result of the efforts to achieve a minimized sonic boom via source disturbance, or N-wave, minimization techniques. A comparison of the D3.2-5 planform with the D3.2-3A baseline is shown in Figure 3-3. The D3.2-5 concept attempts to reduce the sonic boom levels by stretching the lifting length and smoothing the volume and lift distributions.

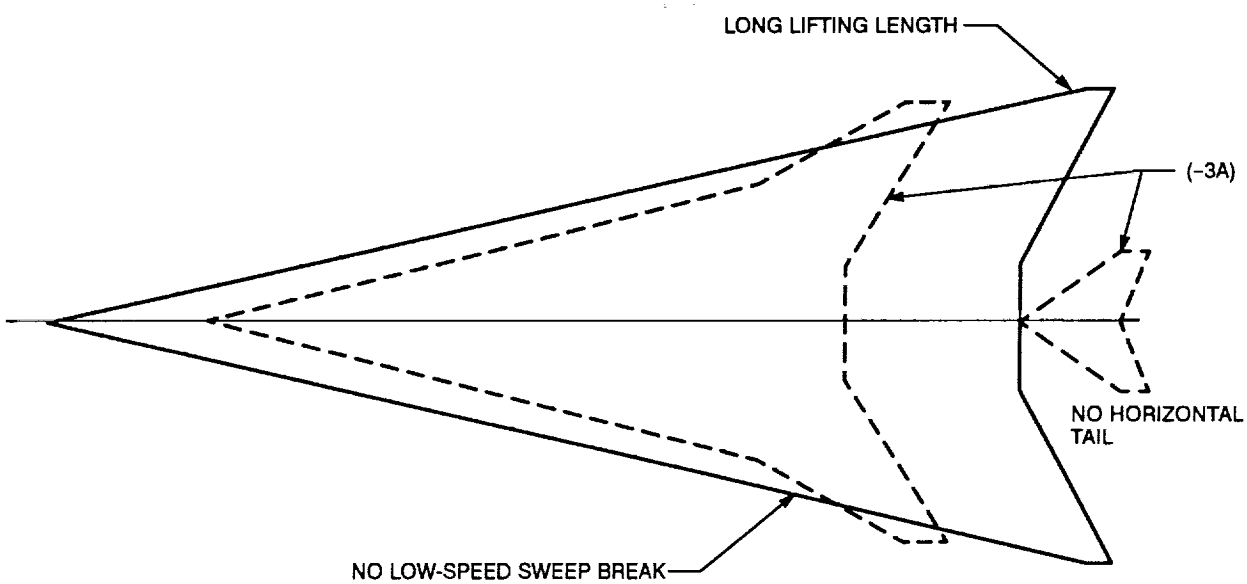


FIGURE 3-3. COMPARISON OF -5 AND -3A PLANFORMS

3.3.2 Wing Design

The exposed lifting length of the D3.2-5 aircraft was increased to 303 feet, and the fuselage length was increased to 325 feet. This represents a 62-percent increase in lifting length over the D3.2-3A configuration and is an important factor in N-wave minimization. The D3.2-3A and D3.2-5 wing characteristics are compared in Table 3-1.

TABLE 3-1
COMPARISON OF D3.2-3A AND D3.2-5 WING CHARACTERISTICS

	3.2-3A	3.2-5
WING AREA (FT^2)	9,500	18,876
LEADING EDGE SWEEP (DEG)	76/62	77.7
T/C	0.0235	0.0235
Æ	1.547	0.912
TRAILING EDGE SWEEP (DEG)	0/-35	0/-32

LRC005-A18

The leading edge sweep breaks in the D3.2-3A configuration cause nonlinearities in the lift distribution at cruise that adversely affect the sonic boom levels. These breaks were removed for the D3.2-5 concept to provide a more linear lift distribution. The wing camber also plays an important role in determining the shape of the lift distribution. The center of wing pressure location and wing camber were also optimized for low-boom considerations. The high-speed performance of the D3.2-5 concept is superior to the D3.2-3A baseline with a trimmed lift-to-drag ratio (L/D) at cruise of 10.5 compared to 9.3, both with full-chord wing (up to flap hinge-line) LFC. A plot of L/D versus Mach number and the axial lift distribution for the D3.2-5 aircraft are shown in Figures 3-4 and 3-5, respectively.

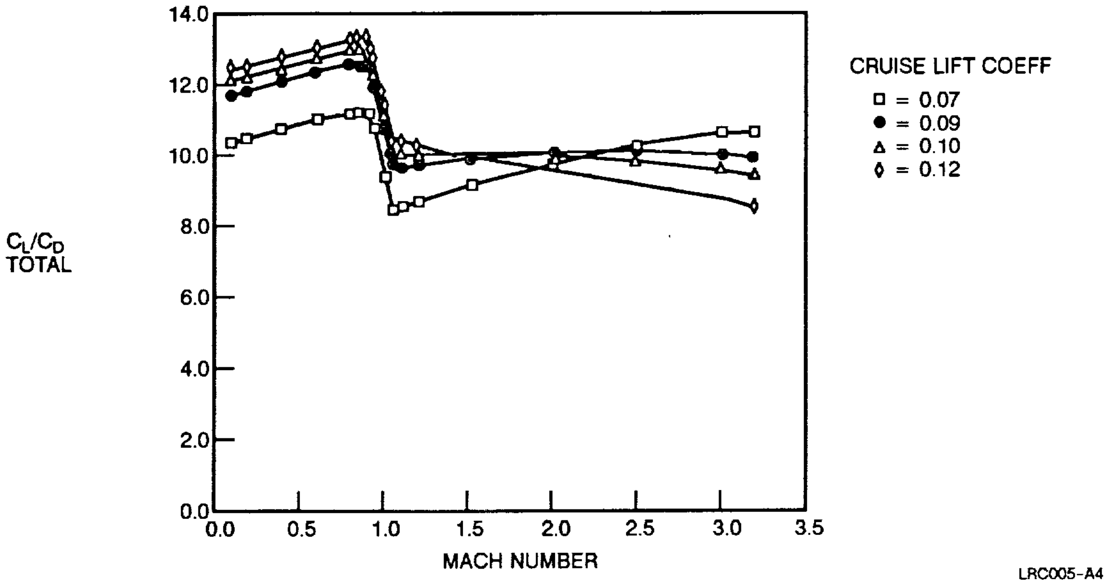


FIGURE 3-4. LIFT-TO-DRAG RATIO VERSUS MACH NUMBER FOR D3.2-5 CONFIGURATION

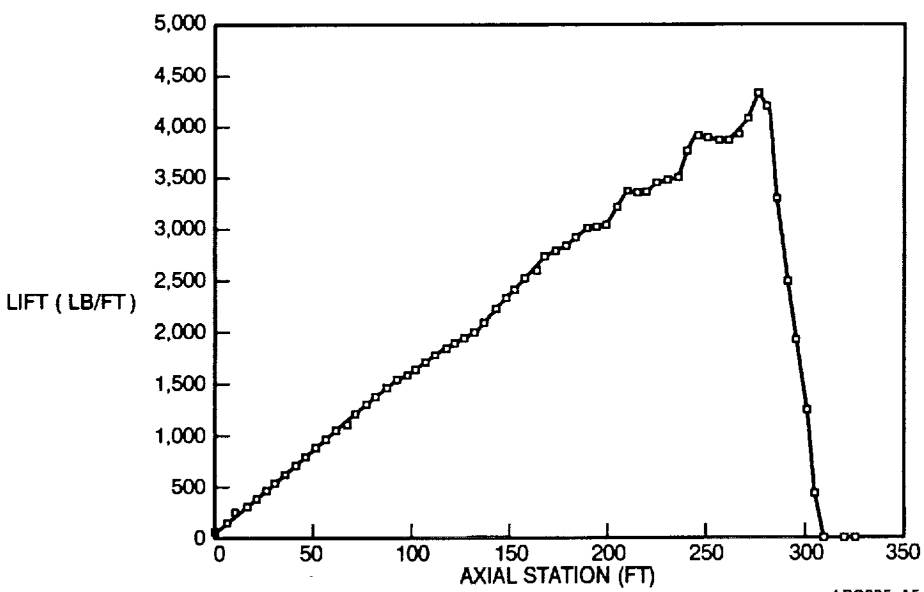


FIGURE 3-5. LIFT DISTRIBUTION FOR THE D3.2-5

Several derivative planforms of the D3.2-5 were investigated in an attempt to improve low-speed aerodynamics and sonic boom levels. Several of these concepts are shown in Figure 3-6. None of these configurations were able to improve on the overall performance of the D3.2-5 concept, primarily because they lacked the large wing area required to achieve acceptable low-speed performance. The results indicate that the optimal design for N-wave minimization is a highly swept wing with a straight leading edge just inside the Mach cone. The wing should start at the aircraft nose (or possibly in front of the fuselage) and run all the way back to the tail to allow for the largest possible lifting length.

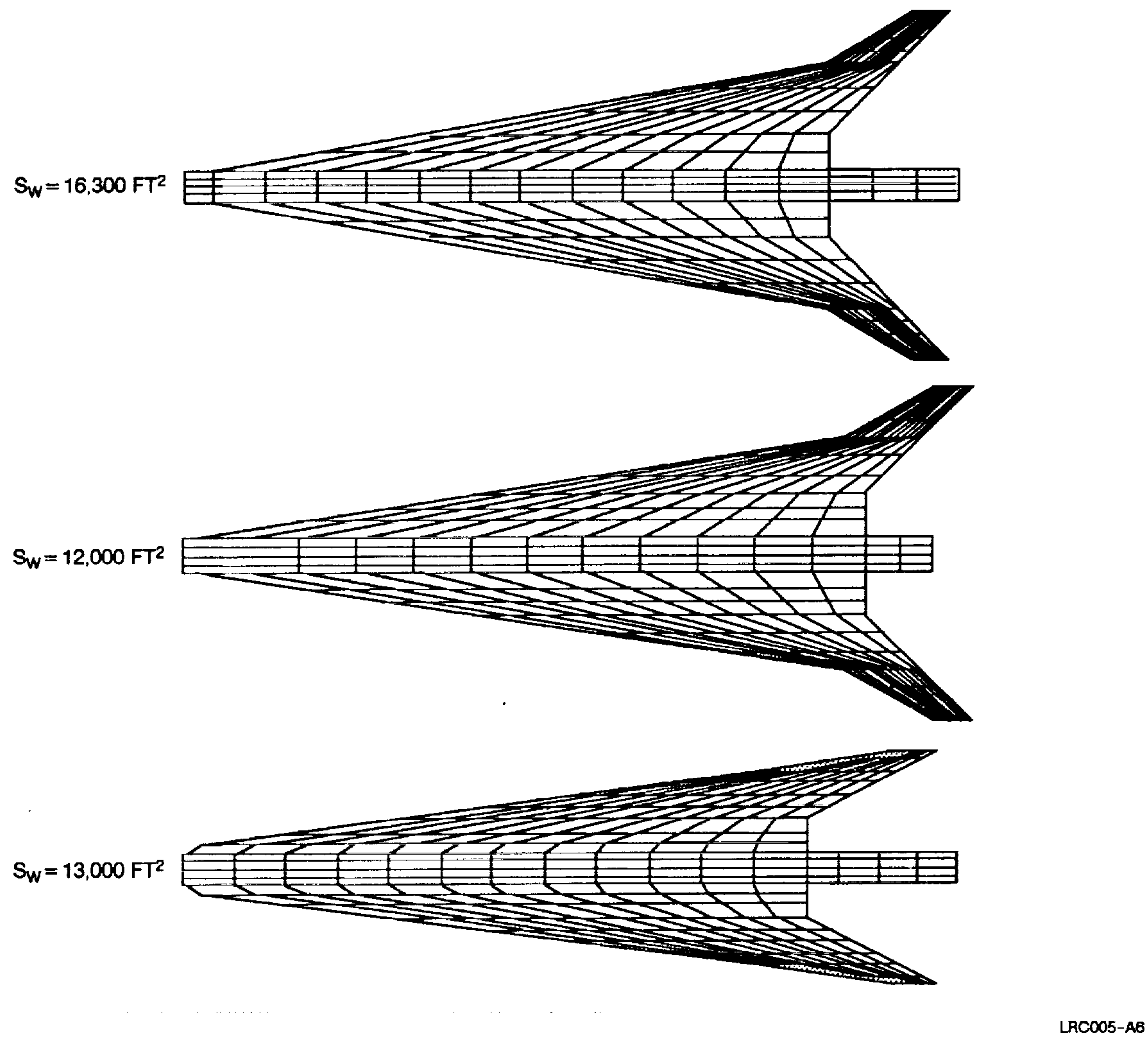


FIGURE 3-6. DERIVATIVE PLANFORM CONCEPTS OF THE D3.2-5 CONFIGURATION

3.3.3 Low-Speed Aerodynamics

Removing the low-speed sweep break lowers the low-speed $(L/D)_{max}$ of the D3.2-5 concept slightly from the D3.2-3A level. The $(C_L)_{max}$ is also reduced, but this is compensated for by a 99-percent increase in wing area and, as a result, the low-speed characteristics of the D3.2-3A and D3.2-5 concepts are very similar. These characteristics are shown in Table 3-2. A small, stowable, low-speed canard was initially added to improve the low-speed aerodynamics of the D3.2-5 concept, although subsequent analysis showed it to unnecessary. However, it is included in the performance and weight analyses for this study.

TABLE 3-2
COMPARISON OF LOW-SPEED CHARACTERISTICS FOR D3.2-3A AND D3.2-5 CONFIGURATIONS

	D3.2-3A	D3.2-5
Sref (FT^2)	9,500	18,650
L/D MAX	17.5	13.0
L/D AT APPROACH CLMax	7.0	7.0
L/D AT TAKEOFF CLMax	7.0	7.0
TAKEOFF CLMax	0.45	0.28

LRC005-A19

3.3.4 Stability and Control

The D3.2-5 configuration is controlled by wing trailing edge elevons and vertical tail rudders. The elevons trim the aircraft at low speed with a small drag penalty at low C_L's, and a slight drag reduction at takeoff and landing C_L maximums. A full-span leading edge flap is used to reduce low-speed drag and pitch-up instability.

3.3.5 Performance and Weight Assessment

The weight analysis for the D3.2-5 concept includes conceptual design weight methodology, advanced aluminum metal matrix composite structures, and advanced aircraft systems concepts consistent with the D3.2-3A baseline concept.

The structure weight includes the wing, fuselage, and vertical tail. The power plant, which is based on the P&W STF905 baseline engine, includes the variable-geometry bicone inlet structure and system, nacelle, functional engine systems, mounting structure, bare engine, nozzle, acoustic suppressor, and thrust reverser. All of the required aircraft subsystems are included in the weight analysis, including a full wing chord laminar flow control system.

The D3.2-5 concept was sized to meet the mission requirements of a 6,500-nautical-mile range and an 11,000-foot takeoff field length for an ISA plus 10°C reference day. The sized D3.2-5 sonic boom concept geometry and weight data are shown in Table 3-3. The results of the performance analysis are shown in Table 3-4. Flight conditions at the beginning of cruise for the sonic boom analysis were a weight of 714,000 pounds and an altitude of 69,100 feet.

3.3.6 Sonic Boom Results

The D3.2-5 configuration achieved a reasonable degree of sonic boom reduction over the D3.2-3A baseline levels, but did not meet the interim goals set for the study. A comparison of the sonic boom waveforms and levels for the two aircraft is shown in Figure 3-7. At the beginning of cruise, the D3.2-5 waveform has a front shock overpressure of 1.4 psf, with a small sawtooth to 1.7 psf. This translates into a loudness level of 98.8 PLdB, a 3.2-PLdB improvement over the D3.2-3A level.*

It is unlikely that any further significant sonic boom minimization is available through aerodynamic refinement of the D3.2-5 vehicle. In order to further reduce the boom levels it will be necessary to

* It should be noted that the beginning of cruise is likely to be the most critical regime for sonic boom because the aircraft is at its heaviest weight and lowest altitude. (The climb-out boom may, in fact, be more critical than the beginning-of-cruise boom, but was not investigated in this study because of the relatively small area affected. Future studies, however, will address this regime as well.) Therefore, unless otherwise noted, all sonic boom levels presented in this study are for the beginning of cruise, directly underneath the aircraft flight track.

TABLE 3-3.
D3.2-5 CONCEPT GEOMETRY AND WEIGHT DATA

GEOMETRY DATA	
MACH NUMBER	3.2
RANGE (N MI)	6,500
FUEL TYPE	TSJF
NUMBER OF PASSENGERS	300
TAKEOFF GROSS WEIGHT (LB)	822,600
MAXIMUM ZERO FUEL WEIGHT (LB)	361,700
MAXIMUM SPACE-LIMITED PAYLOAD (LB)	66,500
WING AREA - TOTAL PLANFORM (FT^2)	18,876
STOWABLE FRONT CANARD - TOTAL PLANFORM (FT^2)	700
VERTICAL TAIL AREA - TOTAL PLANFORM (FT^2)	1,310
NUMBER OF ENGINES	4
MAXIMUM SEA LEVEL STATIC DRY THRUST PER ENGINE (LB)	29,300
WEIGHT DATA (LB)	
STRUCTURES	132,645
POWER PLANT	34,875
SYSTEMS	120,600
MANUFACTURER'S EMPTY WEIGHT	288,120
OPERATOR ITEMS	7,100
PAYLOAD	61,500
ZERO FUEL WEIGHT	356,700
FUEL	465,880
TAKEOFF GROSS WEIGHT	822,600

LRC005-A20

TABLE 3-4
PERFORMANCE RESULTS FOR D3.2-5 CONFIGURATIONS

RANGE (N MI)	6,500
TOGW (LB)	822,600
OEW (LB)	295,300
THRUST (LB, SEA LEVEL STATIC)	61,100
FUEL BURN (LB)	404,800

LRC005-A21

reduce aircraft weight considerably. It does not appear, however, that minimizing N-wave will result in an acceptable sonic boom configuration for a Mach 3.2, 6,500-nautical-mile aircraft for a year 2000 to 2010 certification date.

3.4 WAVEFORM SHAPING

Achieving a minimized, shaped, sonic boom waveform on the ground requires careful design of the aircraft. This is particularly true at Mach 3.2, where the tendency of the pressure signature to degenerate into its asymptotic N-wave form is very strong. There is no closed-form solution to the problem and, as a result, the design process relies heavily on iterative methods.

The first step in the design process is to make some assumptions concerning aircraft characteristics. This is required to generate a target equivalent area distribution that may be used as the design goal. For this study, the assumptions were taken directly from the D3.2-3A configuration and are as follows:

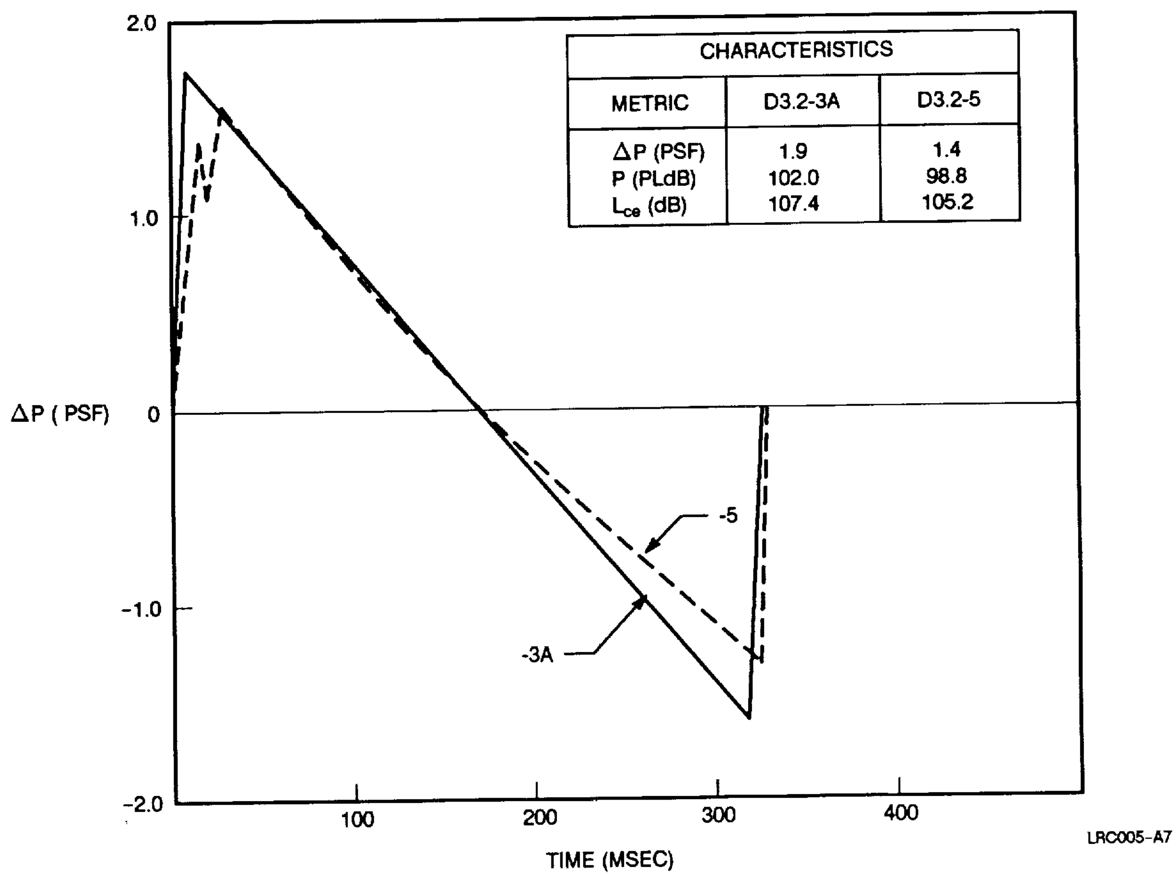


FIGURE 3-7. COMPARISON OF -5 AND -3A WAVEFORMS

- Mach number = 3.2
- Beginning of cruise weight = 669,000 pounds
- Beginning of cruise altitude = 65,000 feet
- Length = 315 feet

The equivalent area distribution associated with these parameters from SEEB is shown in Figure 3-8. The equivalent area of the D3.2-3A baseline is also shown for comparative purposes. The sonic boom waveform associated with the target equivalent area is shown in Figure 3-9.

3.4.1 Wing Design

In Figure 3-8, it can be seen that a large amount of equivalent area relative to the D3.2-3A is needed at the nose of the aircraft to match the target equivalent area. This can be accomplished by adding volume (bluntness) to the nose or by adding lift. Adding volume to the nose increases drag and deteriorates aircraft performance; adding lift presents a challenging design integration problem. The new configuration, the D3.2-12, uses lift to add equivalent area forward and attempts to minimize any drag penalty at cruise.

A comparison of the D3.2-12 and D3.2-3A planforms is shown in Figure 3-10. The primary feature of the D3.2-12 is the large front canard. The canard adds the appropriate amount of equivalent area needed in the forward section of the configuration to avoid shock coalescence. Other sonic boom

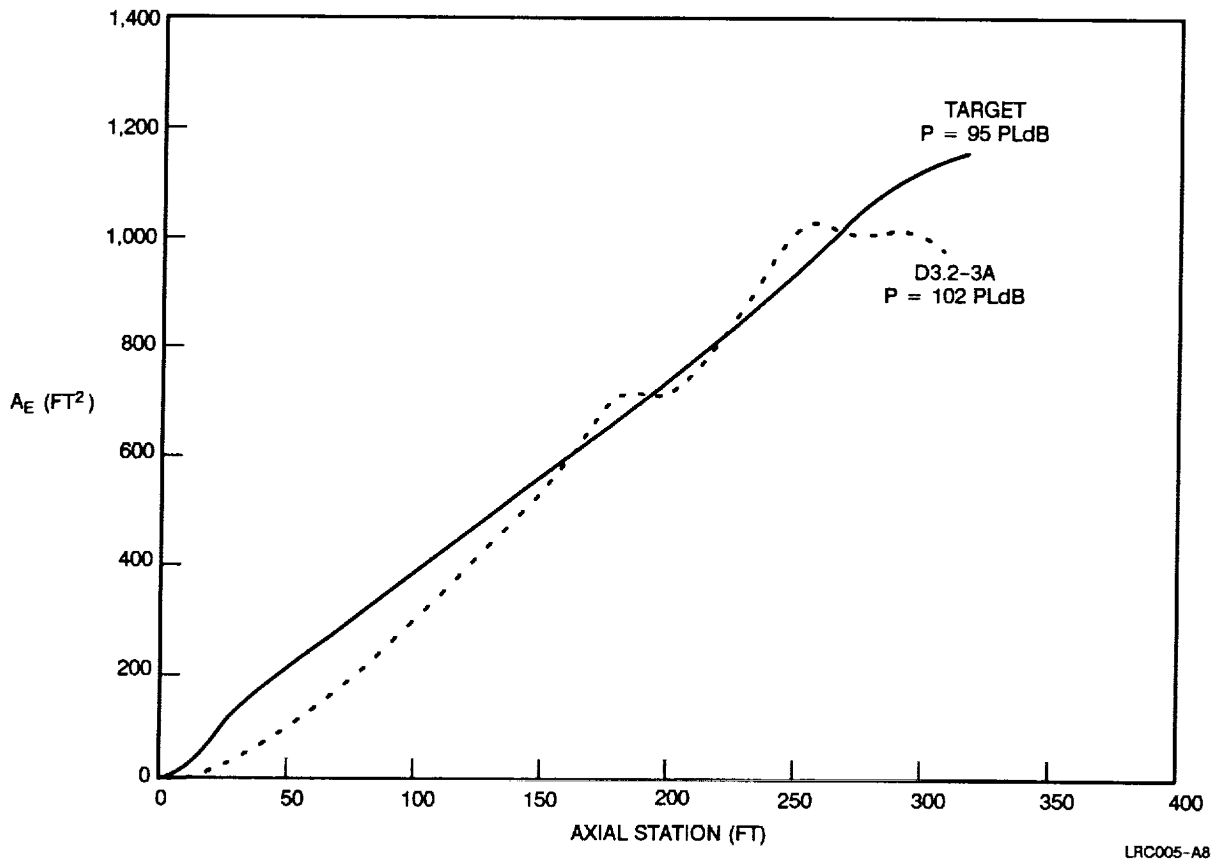


FIGURE 3-8. EQUIVALENT AREA COMPARISON OF D3.2-3A AND TARGET

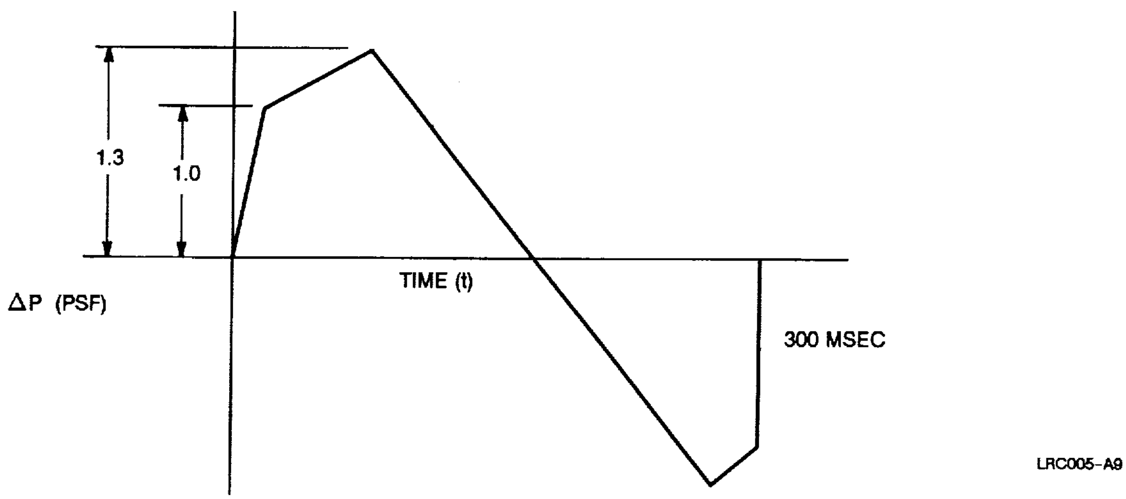


FIGURE 3-9. SONIC BOOM SHAPING TARGET WAVEFORM

features include a long lifting length and highly swept main wing. A comparison of the D3.2-3A and D3.2-12 wing characteristics is shown in Table 3-5.

The effect of the canard downwash on the performance of the main wing was one of the aerodynamic concerns for the D3.2-12. The configuration was analyzed both with and without the canard. Results obtained through the linear supersonic theory showed that the canard does not have a significant affect on the main wing at cruise. Compared to the wing-alone configuration, adding the canard

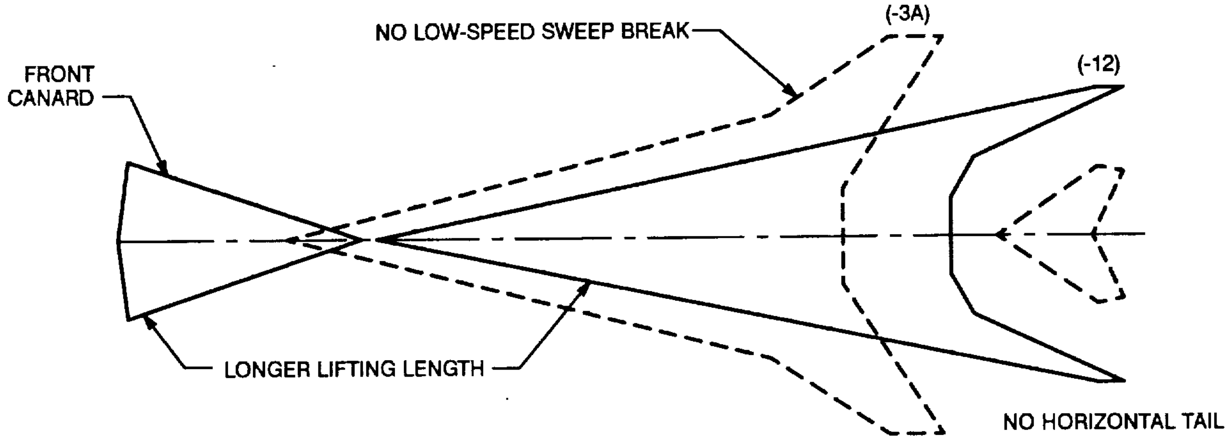


FIGURE 3-10. COMPARISON OF D3.2 -12 AND D3.2-3A PLANFORMS

TABLE 3-5
COMPARISON OF D3.2-3A AND D3.2-12 WING CHARACTERISTICS

	D3.2-3A	D3.2-12
WING AREA (FT^2)	9,500	WING: 8,439 CANARD: 1,938
LEADING EDGE SWEEP (DEG)	76/62	78.6
TRAILING EDGE SWEEP (DEG)	0/-35	0/-30/-68
τ/C MAX	0.0235	0.0235
Æ	1.547	1.088

LRC005-A22

decreases the L/D at cruise by about 1.0 percent. The fact that the penalty is small can be attributed to the large vertical height difference between the wing and canard (7.6 feet), the high sweep angles of the canard trailing edge and wing leading edge, and the large axial distance between the canard and wing. The L/D of the D3.2-12 versus Mach number is shown in Figure 3-11.

The canard is flown at a 3-degree angle of attack relative to the wing in order to generate the lift required to shape the boom. This has implications on the trimming of the aircraft and the center of gravity (CG) location. In order to trim the aircraft at cruise, the engine nozzles are vectored down 8 degrees. The camber of the wing was optimized for the wing-alone condition, and then the canard was added as a flat plate. The lift distribution of the D3.2-12 is shown in Figure 3-12.

3.4.2 Low-Speed Aerodynamics

The main wing of the D3.2-12 configuration has a poor low-speed performance because of its relatively small wing area (8,439 ft^2), low aspect ratio, and high sweep angle. The theoretical $(L/D)_{max}$ of the D3.2-12 compares well to the D3.2-3A, but the $(C_L)_{max}$ is low. To compensate for the low $(C_L)_{max}$ of the aircraft, a high flap setting (30 degrees) is required. This reduces the L/D at takeoff and approach to 5.3. A comparison of the low-speed performance of the D3.2-12 and D3.2-3A is shown in Table 3-6.

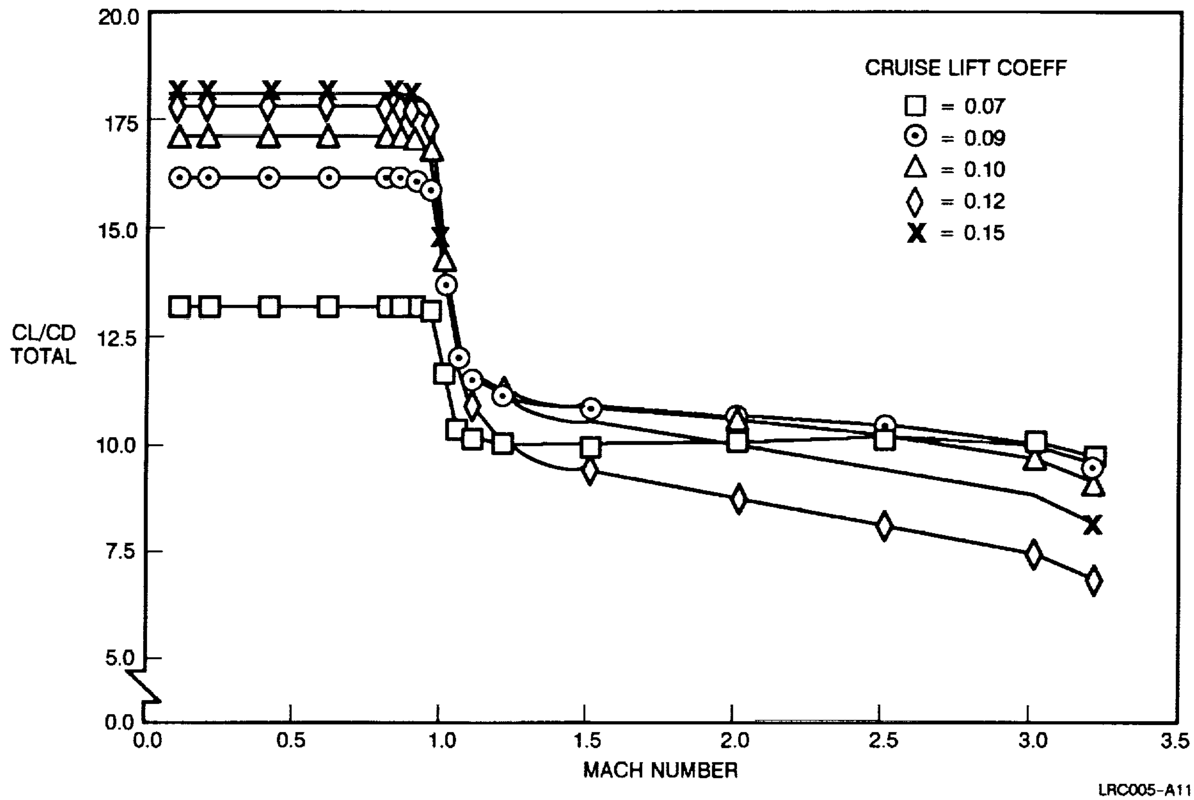


FIGURE 3-11. LIFT-TO-DRAG RATIO VERSUS MACH NUMBER FOR D3.2-12

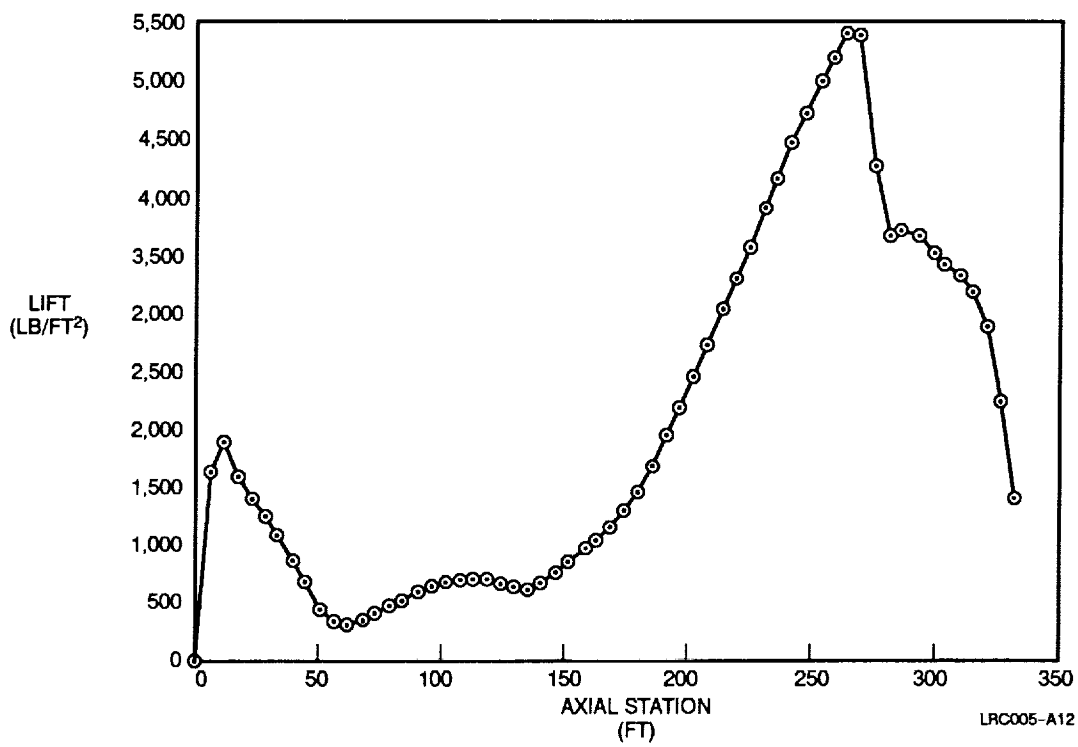


FIGURE 3-12. LIFT DISTRIBUTION OF D3.2-12

TABLE 3-6
COMPARISON OF LOW-SPEED CHARACTERISTICS FOR D3.2-3A AND D3.2-12 CONFIGURATIONS

	D3.2-3A	D3.2-12
Sref (FT^2)	9,500	9,500
L/D MAX	17.5	17.5
L/D AT APPROACH CLMax	7.0	5.3
L/D AT TAKEOFF CLMax	7.0	5.3
TAKEOFF CLMax	0.45	0.35

LRC005-A23

Low-speed improvements to the D3.2-12 can be realized by increasing the wing area and/or increasing the wing aspect ratio. Low-speed improvements without incurring sonic boom penalties would be the next objectives.

3.4.3 Stability and Control

Trim for the D3.2-12 is achieved through CG management and canard control surface deflection. With optimization, the D3.2-12 can be trimmed with a small L/D benefit. This is accomplished by keeping the canard lift to a minimum through moving the CG to the aft stability limit (–10-percent static stability). Pitch control is maintained with leading and trailing edge control surfaces on the canard. Full-span leading and trailing edge flaps are employed with outboard flaperons.

3.4.4 Performance and Weight Assessment

The advanced aluminum metal matrix composite structures, advanced aircraft systems concepts, power plant, and systems weight definitions are similar to the D3.2-5. The D3.2-12 was sized to meet the mission range requirement of 6,500 nautical miles. The sized D3.2-12 sonic boom concept geometry and weight data are shown in Table 3-7. The results of the performance analysis are shown in Table 3-8. Flight conditions at the beginning of cruise for sonic boom analysis were a weight of 616,000 pounds and an altitude of 65,800 feet.

3.4.5 Sonic Boom Results

The 96.5-PLdB loudness level achieved by the D3.2-12 represents a 5.5-PLdB reduction compared to the D3.2-3A. Sonic boom results of the D3.2-12 and D3.2-3A are shown in Figure 3-13. The pressure signature of the D3.2-12 is characterized by multiple shocks in the positive pressure region and a very small aft shock. The aft shock prediction is somewhat uncertain because of difficulties in modeling the aft portion of the aircraft.

The target and D3.2-12 equivalent area distributions are compared in Figure 3-14, which shows the source of the double shock to be a dip in equivalent area between the canard and main wing. This can be remedied with a chine between the canard and wing. However, higher order design methodologies must be developed before a chined configuration can be analyzed.

3.5 SENSITIVITY STUDIES

Sensitivity studies of the influence of overland Mach number and design range on aircraft sonic boom characteristics were carried out using the D3.2-3A configuration. In order to gain an understanding of how the overland Mach number affects sonic boom levels, a sample mission was created.

3.5.1 Overland Mach Number

The sample mission consists of 1,500 nautical miles overland followed by 5,000 nautical miles overwater, for a total range of 6,500 nautical miles. For the overland portion of the mission, the D3.2-3A

TABLE 3-7

D3.2-12 CONCEPT GEOMETRY AND WEIGHT DATA

GEOMETRY DATA	
MACH NUMBER	3.2
RANGE (N MI)	6,500
FUEL TYPE	TSJF
NUMBER OF PASSENGERS	300
TAKEOFF GROSS WEIGHT (LB)	697,200
MAXIMUM ZERO FUEL WEIGHT (LB)	309,000
MAXIMUM SPACE-LIMITED PAYLOAD (LB)	66,500
WING AREA - TOTAL PLANFORM (FT^2)	9,500
STOWABLE FRONT CANARD - TOTAL PLANFORM (FT^2)	2,314
VERTICAL TAIL AREA - TOTAL PLANFORM (FT^2)	768
NUMBER OF ENGINES	4
MAXIMUM SEA LEVEL STATIC DRY THRUST PER ENGINE (LB)	26,800
WEIGHT DATA (LB)	
STRUCTURES	96,490
POWER PLANT	31,900
SYSTEMS	107,020
MANUFACTURER'S EMPTY WEIGHT	235,410
OPERATOR ITEMS	7,100
PAYLOAD	61,500
ZERO FUEL WEIGHT	304,010
FUEL	393,170
TAKEOFF GROSS WEIGHT	697,200

LRC005-A24

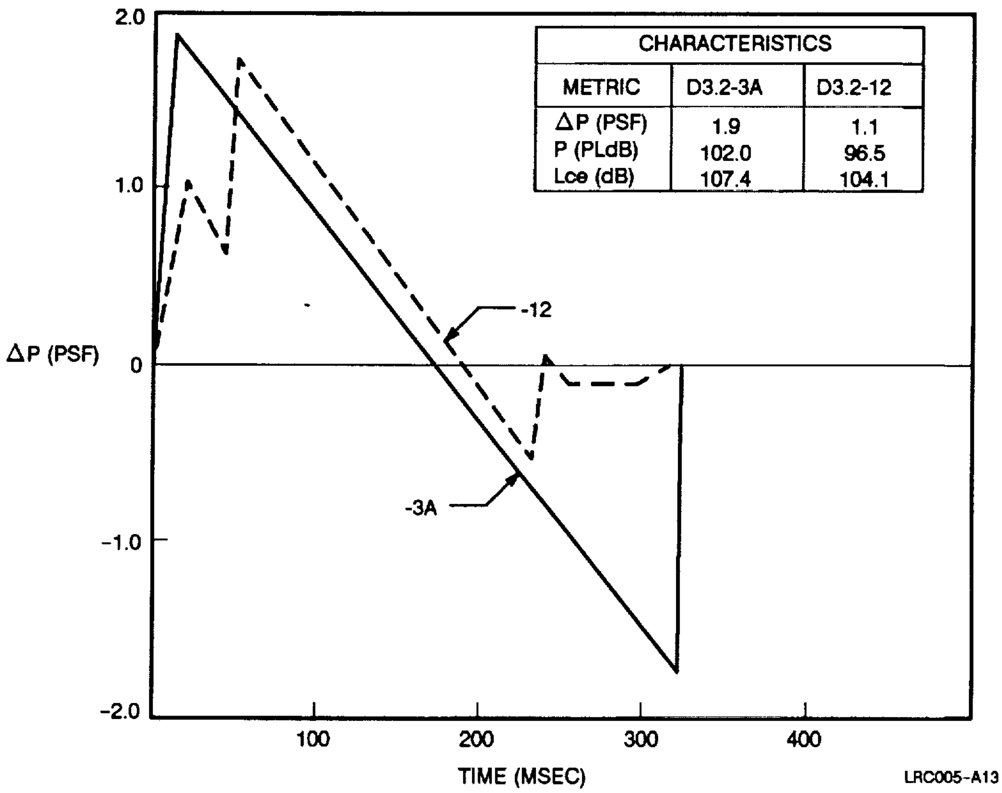
TABLE 3-8

PERFORMANCE RESULTS FOR D3.2-12 CONFIGURATION

RANGE (N MI)	6,500
TOGW (LB)	697,200
OEW (LB)	241,300
THRUST (LB, SEA LEVEL STATIC)	55,880
FUEL BURN (LB)	351,900

LRC005-A25

was flown at three different Mach numbers (1.5, 2.4, and 3.2). The remainder of the mission (overwater) was flown at the aircraft design cruise speed of Mach 3.2. The three cases were flown with a fixed TOGW of 769,000 pounds. Aircraft performance was considered in such a way that the altitudes and weights of the three cases were consistent, but the overall range was allowed to vary. A comparison of the overpressure and perceived loudness for the three cases over the mission range is shown in Figures 3-15 and 3-16. These figures present two different scenarios and indicate the importance of carefully selecting the appropriate metric for sonic boom assessment. The lowest overpressures are obtained by cruising overland at Mach 1.5 as indicated in Figure 3-15. If overpressure was the only figure of merit, it would be concluded that lower Mach numbers in general and Mach 1.5 in particular allow for lower sonic boom levels than Mach 3.2. This is not the case, however. A comparison of the Mach 1.5 and Mach 3.2 waveforms during overland cruise is shown in Figure 3-17. The Mach 1.5 waveform has three distinct shocks in the positive overpressure region, while the shocks in the Mach 3.2 waveform have coalesced into a basic N-wave. When the loudness levels of the two



CHARACTERISTICS		
METRIC	D3.2-3A	D3.2-12
ΔP (PSF)	1.9	1.1
P (PLdB)	102.0	96.5
Lce (dB)	107.4	104.1

FIGURE 3-13. COMPARISON OF -12 AND -3A WAVEFORMS

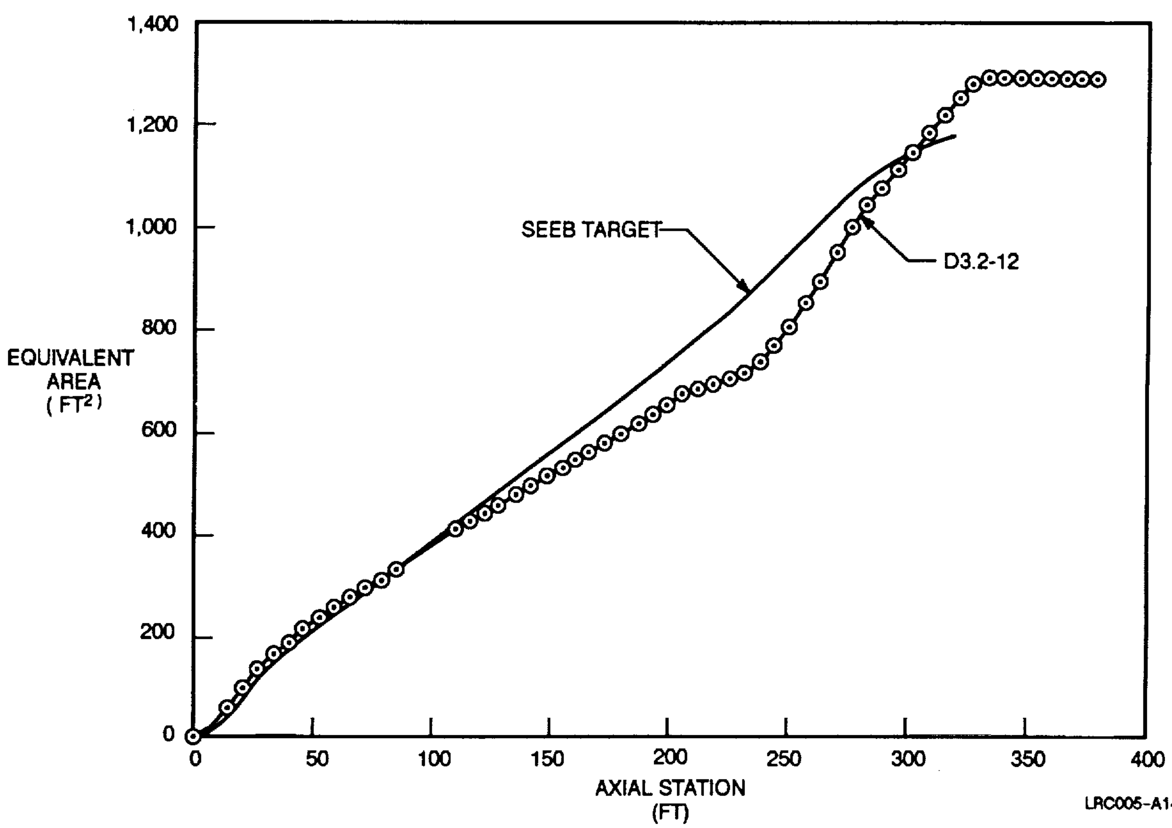


FIGURE 3-14. COMPARISON OF D3.2-12 EQUIVALENT AREA TO TARGET

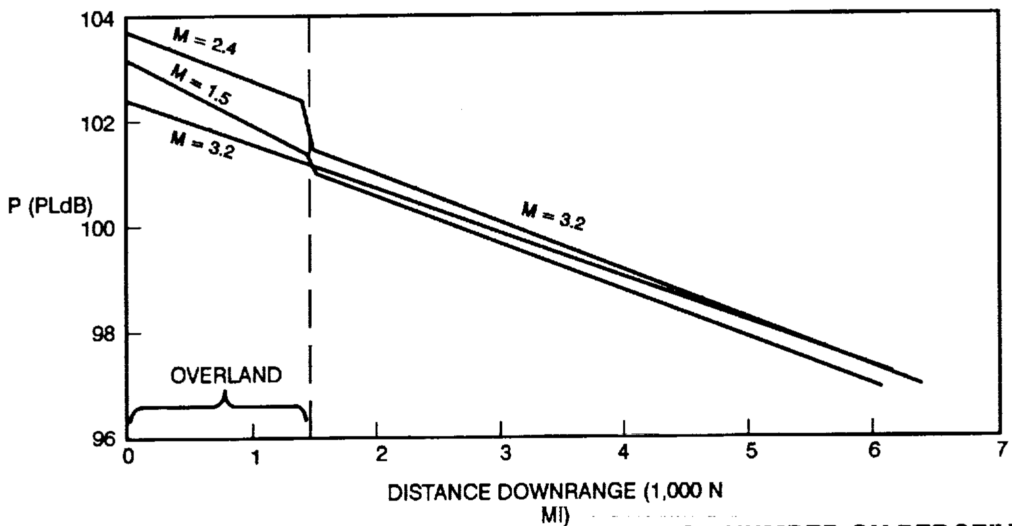


FIGURE 3-15. EFFECT OF OVERLAND MACH NUMBER ON PERCEIVED LEVEL FOR D3.2-3A CONFIGURATION

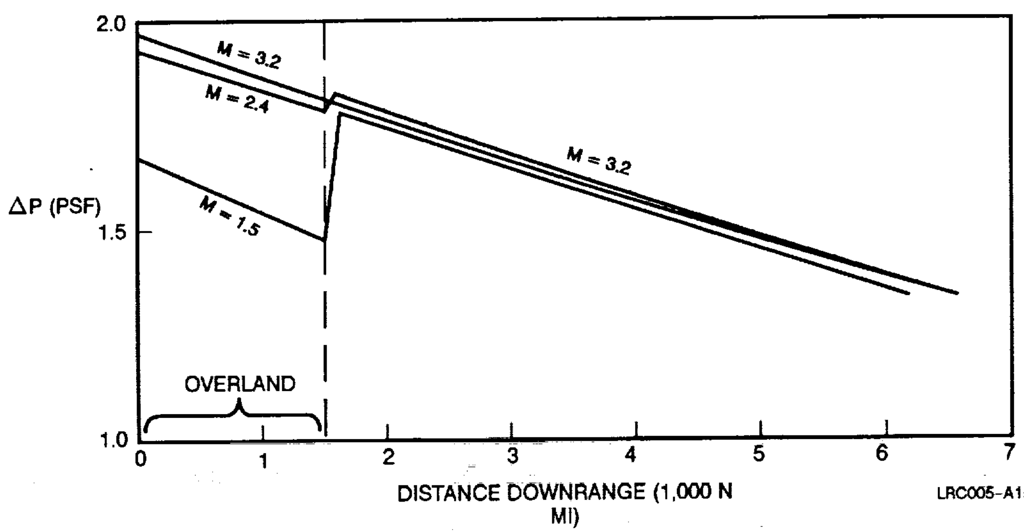


FIGURE 3-16. EFFECT OF OVERLAND MACH NUMBER ON BOOM OVERPRESSURE FOR D3.2-3A CONFIGURATION

waveforms are calculated, the results indicate that the three shocks in the Mach 1.5 waveform would be perceived by the ear to be louder than the single, stronger shock of the Mach 3.2 waveform.

By flying lower and slower, shock coalescence was avoided, but no favorable trade was found for perceived loudness. Therefore, it can be concluded in general that aircraft speed (so long as it is supersonic) is not an effective means of improving sonic boom levels. Aircraft speed can, however, be effectively utilized to shape a waveform into other than a basic N-wave. The Mach 1.5 results suggest that a low supersonic Mach number cruise might be used in conjunction with a carefully tailored planform to generate shaped sonic booms of the type proposed by Seebass and George (Reference 3-1).

3.5.2 Design Range

The mission range of the aircraft has a second-order effect on sonic boom levels by sizing the amount of fuel required to make the mission, and thus the takeoff gross weight. It has been speculated that

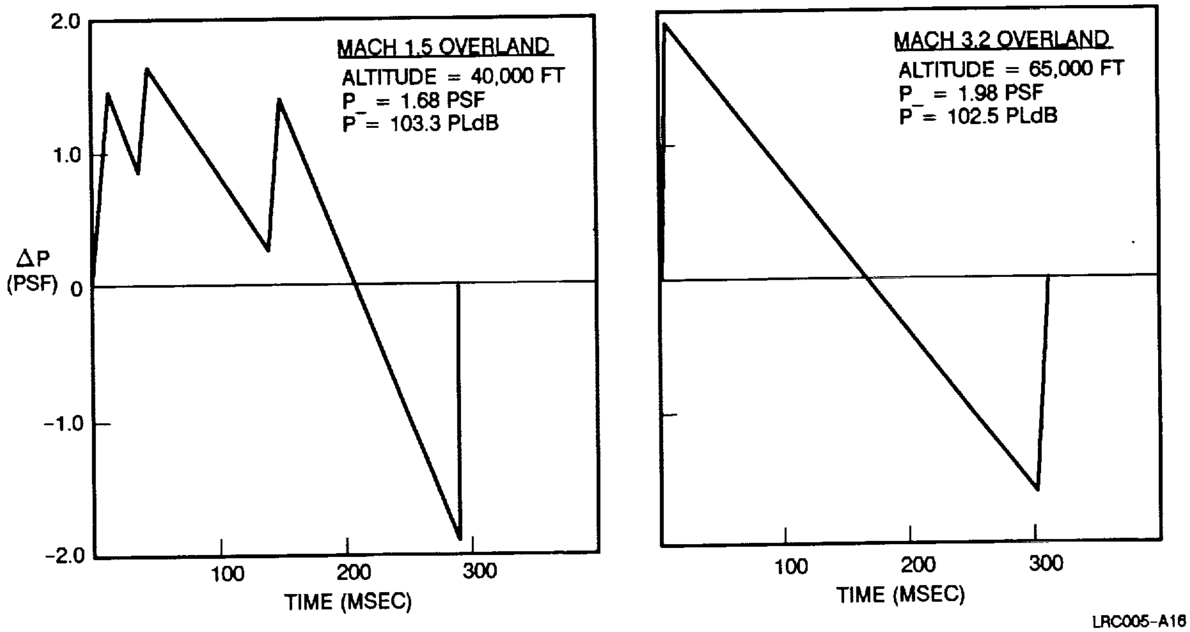


FIGURE 3-17. COMPARISON OF OVERLAND WAVEFORMS FOR 3.2-3A CONFIGURATION

an aircraft designed for a 6,500-nautical-mile range may be able to fly supersonically overland for shorter missions.

The D3.2-3A configuration was used as a representative aircraft for the study. The climb-out and descent flight segments of the baseline 6,500-nautical-mile mission were held constant in terms of both distance traveled and fuel burned. The variation in design range was assumed to affect only the cruise segment.

The mission range was varied from 6,500 to 1,800 nautical miles. Figures 3-18 and 3-19 compare perceived loudness and maximum overpressure versus design range. In both figures there is a discontinuity at about 3,500 nautical miles, where the shape of the waveform drastically changes.

If the higher range values are considered, it is seen that the relationship between the design range and the sonic boom levels is nearly linear with a slope of approximately 0.1 psf/1 PLdB of boom level reduction per 1,000-nautical-mile decrease in range at the beginning of cruise.

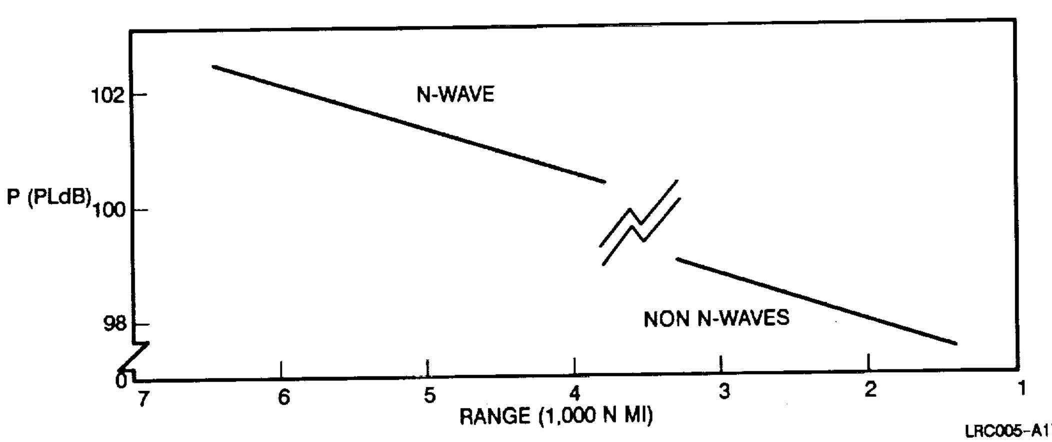


FIGURE 3-18. EFFECT OF MISSION RANGE ON SONIC BOOM LOUDNESS

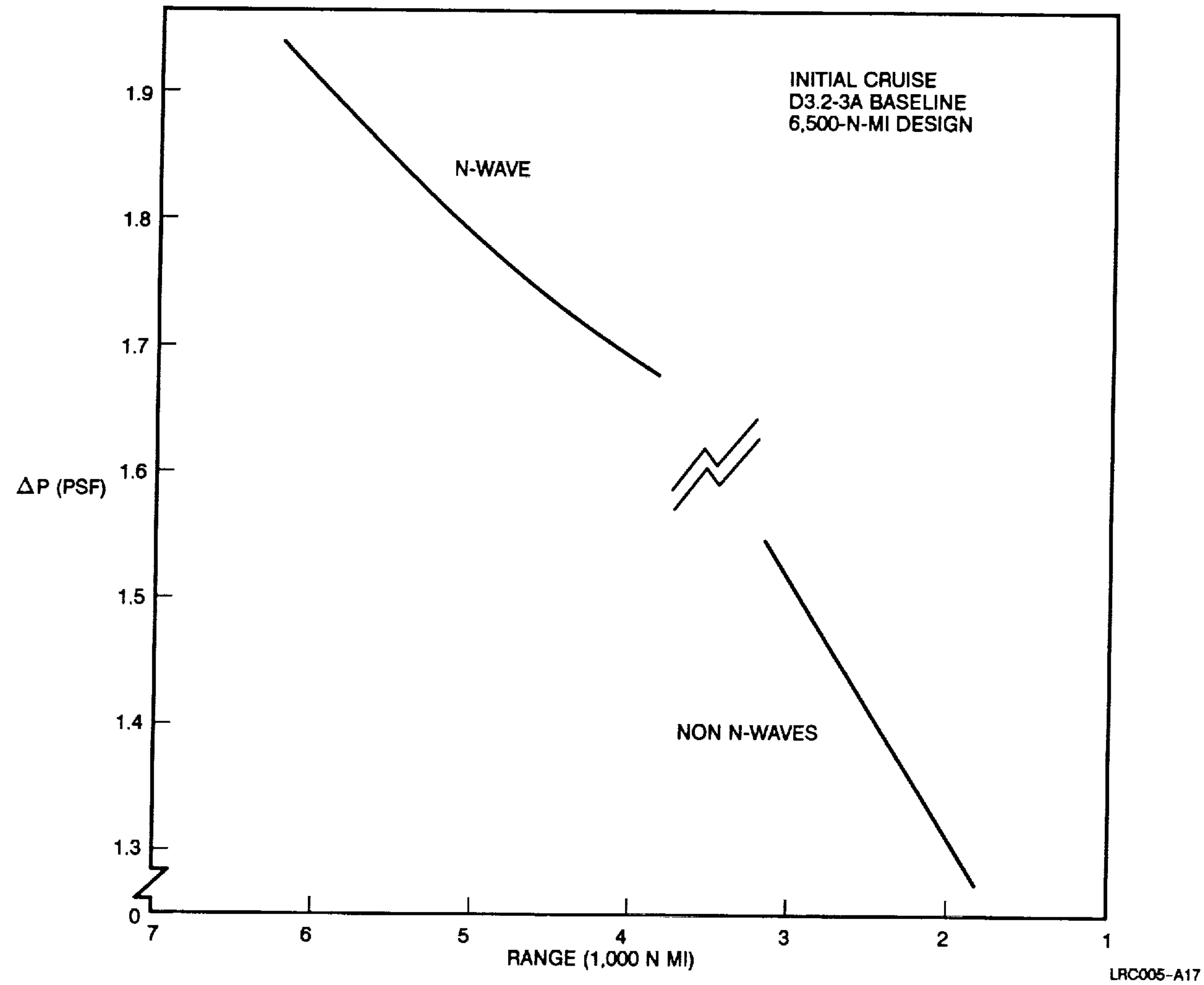


FIGURE 3-19. EFFECT OF RANGE ON SONIC BOOM OVERPRESSURE

3.6 CONCLUSIONS

- The D3.2-3A baseline aircraft generates a sonic boom directly under the flight track at the beginning of cruise with an estimated overpressure of 1.9 psf and loudness level of 102 PLdB. These levels are significantly higher than the tentative acceptable design goals of 0.6 psf and 90 PLdB. The potential for sonic boom minimization through operational techniques such as velocity, altitude, and range control is limited, and operational methods alone will not result in acceptable boom levels for the D3.2-3A.

- Two new configurations were developed for the purpose of reducing sonic boom levels. One of these, the D3.2-5, was an attempt to minimize boom levels via aerodynamic optimization, while the other, the D3.2-12, was an attempt to minimize boom levels by shaping the sonic boom waveform received at ground level. These configurations were designed with the same ground rules and mission requirements as the D3.2-3A baseline. A complete comparison of the three configurations is shown in Table 3-9. The lowest sonic boom levels were attained with the D3.2-12 with a front shock overpressure of 1.1 psf and a perceived loudness of 96.5 PLdB.

- The most feasible way to reduce sonic boom levels at Mach 3.2 for a 6,500-nautical-mile, 300-passenger aircraft is through sonic boom shaping. The approach of obtaining acceptable N-wave levels is not considered to be viable.
- Sonic boom shaping at Mach 3.2 is a difficult task because of the physics of shock coalescence involved. To avoid coalescence, it is necessary to create two very strong shocks, hence the large canard of the D3.2-12.
- The D3.2-12 configuration suggests that waveform shaping at Mach 3.2 is difficult, but not impossible. Two separate shocks are propagated to the ground because of the addition of the front canard. The aerodynamics involved with flying the canard appear to be manageable. The primary weakness of the D3.2-12 is its low-speed performance, and this is a result of the main wing design, not the canard.
- The sonic boom predictions for the D3.2-12 configuration show a relatively small tail shock. However, because of the difficulty in modeling the aft portion of the aircraft with the engine exhaust plumes, the tail shock predictions must be treated with some caution.
- The results of this study are encouraging inasmuch as a 5.5-PLdB reduction in loudness was achieved from the baseline aircraft levels, and further reductions are expected. This study did not address the question of what sonic boom level will be acceptable. It is absolutely imperative that this issue be resolved because all advances in sonic boom minimization, no matter how impressive or noteworthy, will be meaningless unless the ban on overland supersonic commercial flight can be removed.

TABLE 3-9

OVERALL CONFIGURATION COMPARISON

CONFIGURATION	D3.2-3A	D3.2-5	D3.2-12
MTOGW (LB)	769,000	822,600	697,200*
L/D AT CRUISE	8.9	10.5	9.9
L/D AT TAKEOFF	7.0	7.0	5.3*
LOUDNESS AT BEGINNING OF CRUISE (PLdB)	102.0	98.8	96.5

* ASSESSMENTS CONTINUING

LRC005-A26

3.7 RECOMMENDATIONS

- In order to connect the two pressure peaks in the positive overpressure region for the D3.2-12, it is necessary to add a chine between the wing and canard. This was not possible during this study because of the paneling constraints in the linear aerodynamic analysis codes. Higher order methodologies such as CFD are necessary to improve the design process. The application of CFD methods to sonic boom prediction will allow for the assessment of nonstandard planform shapes and the inclusion of nonlinear aerodynamic effects, both of which may be critical for achieving acceptable overland levels.
- More research is needed to understand and be able to model engine exhaust effects; shaping the tail shock may reduce the sonic boom level significantly.
- The physics of waveform shaping should be investigated at lower Mach numbers where altitudes, Mach cones, and shock strengths are more favorable. Also, the effect of aircraft cruise weight with respect to Mach number must be determined.

3.8 REFERENCES

3-1 George, A. R., and Seebass, R. "Sonic Boom Minimization Including Both Front and Rear Shocks," AIAA Journal, 9 (10), pp. 2091-2093, October 1971.

3-2 Darden, C.M., "Sonic-Boom Minimization With Nose-Bluntness Relaxation," NASA TP-1348, 1979.

SECTION 4
EXTERIOR NOISE

4.1 INTRODUCTION

During Phase IIIA, studies have continued with the engine companies to assess HSCT engine cycles and noise suppression hardware that will comply with the current subsonic noise certification rules for new aircraft, FAR Part 36, Stage 3 and ICAO Annex 16, Chapter 3. This noise goal was established during previous HSCT system studies. However, Douglas recognizes two other exterior noise goals that are considered necessary to achieve HSCT environmental acceptance — airport noise and en route noise.

Community noise requirements for HSCTs at international airports should be compatible with the long-range subsonic aircraft that will be operating after the turn of the century. It may be necessary to develop automated noise abatement procedures that will minimize the HSCT airport noise impact. Secondly, it is also recognized that HSCT community noise, below the flight paths during takeoff climb phases to the cruise condition, should be compatible to that created by the existing fleet.

To evaluate the achievement of the above exterior noise goals, two noise reduction concepts have been explored in parallel:

- High-specific-thrust engines with noise-suppression nozzles producing large noise reductions
- High-flow engines with noise-suppression nozzles producing moderate noise reductions

Both GE and P&W supplied engine cycle data and noise suppression hardware information for these concepts. The level of noise suppression used in the noise assessment was jointly determined with the engine companies.

The effects of engine sizing to achieve a 6,500-nautical-mile range with a standard day takeoff field length (TOFL) of 10,600 feet (11,000 feet for an ISA + 10°C acoustic reference day) have been evaluated for the maximum engine power codes (setting) for each engine cycle. All engine configurations have also been evaluated at fixed takeoff weight, landing weight, altitude, and engine thrust conditions to determine the level of noise suppression achieved and shortfalls relative to Stage 3 noise limits.

4.2 ENGINE DATA BASES

Both GE and P&W provided acoustic and performance data for a number of engines. The Phase IIIA engines all assumed a year 1995 engine technology availability date (TAD) corresponding to a year 2005 certification date. The following Mach 3.2 engines were studied:

- GE21/F14, Study M1, augmented variable-cycle engine (VCE) (see Figure 4-1 for engine flow arrangement).
- GE21/FLA1, Study A1, two-stream exhaust, nonaugmented high-flow fan VCE (see Figure 4-2 for engine flow arrangement).
- GE21/FLA1, Study A2, three-stream exhaust, nonaugmented high-flow fan VCE (see Figure 4-3 for engine flow arrangement).
- P&W STF947 augmented variable-stream-control engine (VSCE), both with the baseline convergent-divergent ejector nozzle with chute suppressor and with a high-flow mixer/ejector nozzle (see Figures 4-4 and 4-5, respectively, for engine flow arrangement).

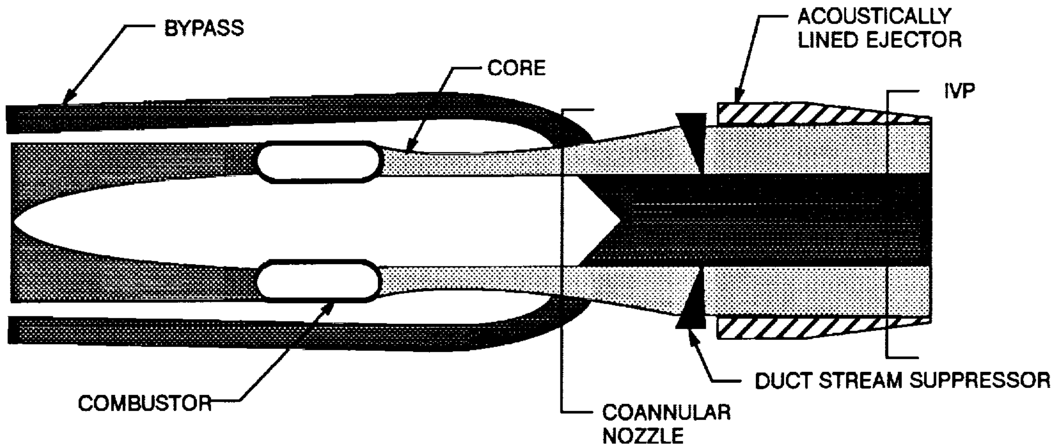


FIGURE 4-1. GENERAL ENGINE FLOW LAYOUT AND SUPPRESSOR HARDWARE - GE-VCE-GE21/F14-STUDY M1

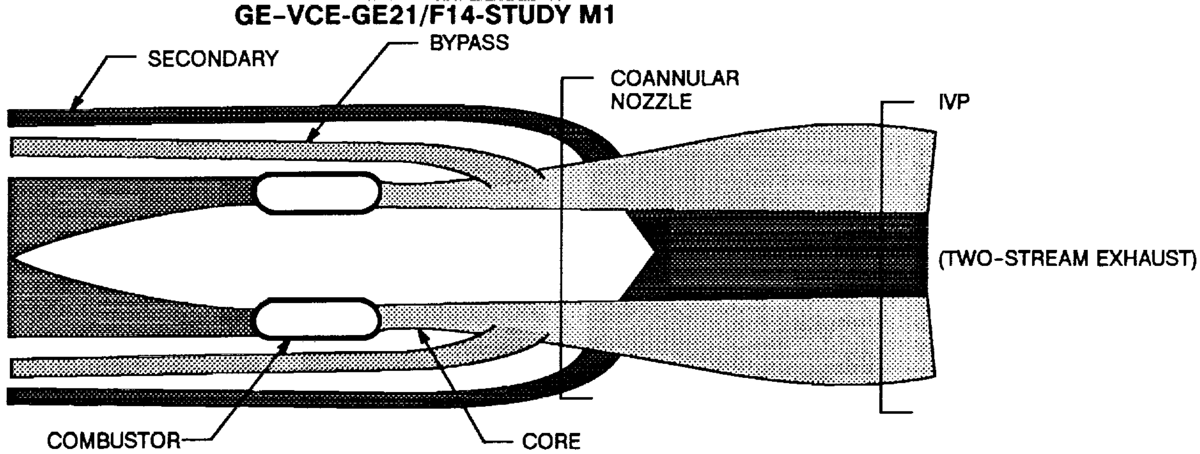


FIGURE 4-2. GENERAL ENGINE FLOW LAYOUT AND SUPPRESSOR HARDWARE - GE-VCE-GE21/FLA1-STUDY A1

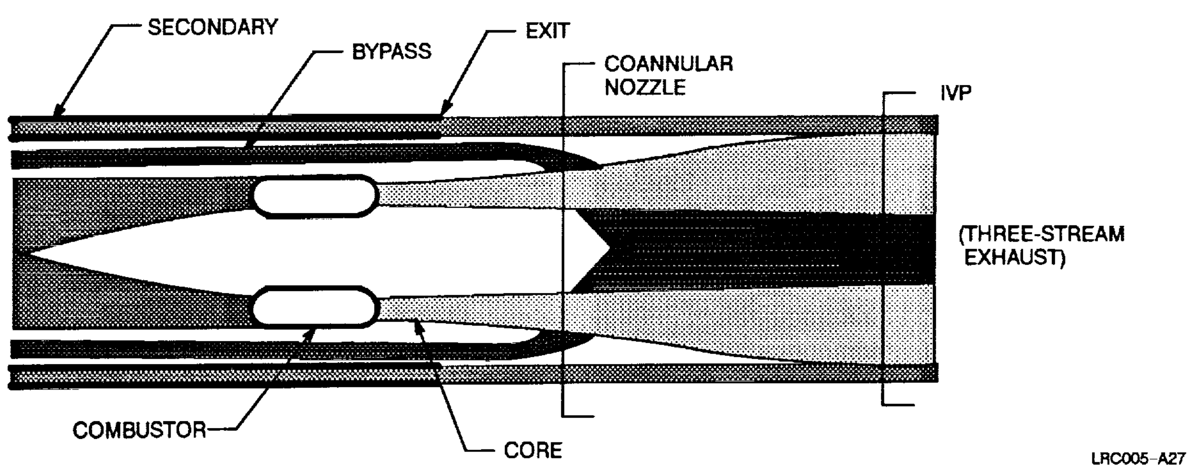


FIGURE 4-3. GENERAL ENGINE FLOW LAYOUT AND SUPPRESSOR HARDWARE - GE-VCE-GE21/FLA1-STUDY A2

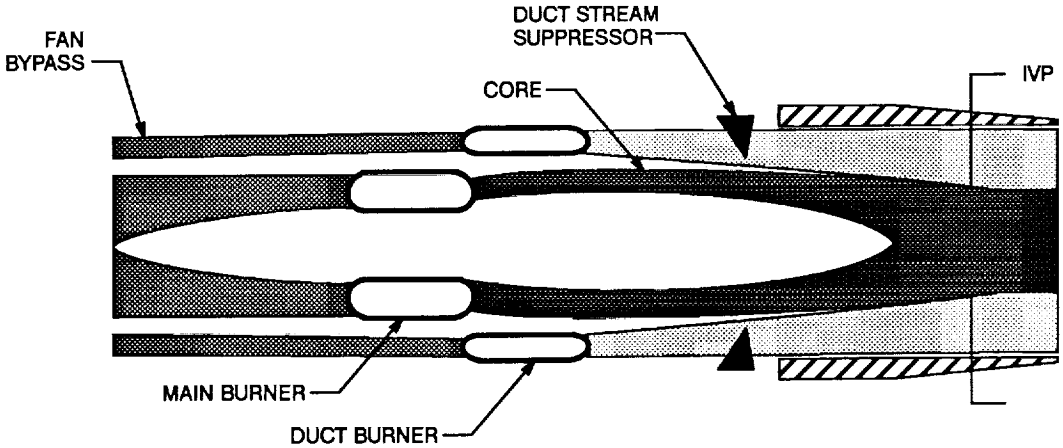


FIGURE 4-4. GENERAL ENGINE FLOW LAYOUT AND SUPPRESSOR HARDWARE - P&W-VSCE-STF947 (BASELINE)

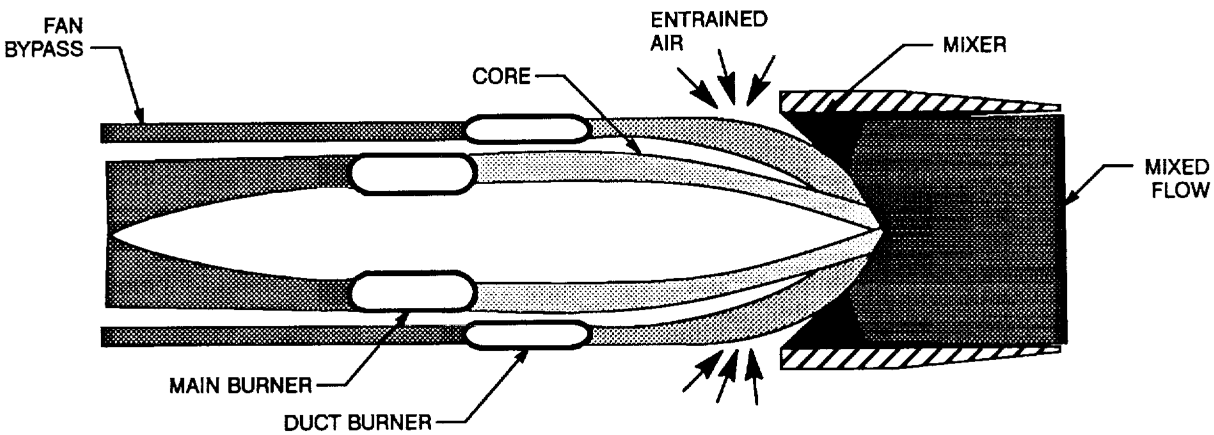
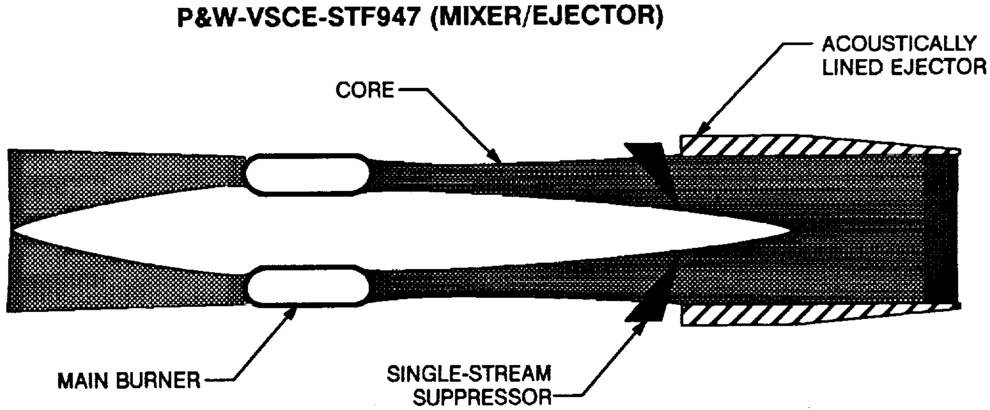


FIGURE 4-5. GENERAL ENGINE FLOW LAYOUT AND SUPPRESSOR HARDWARE - P&W-VSCE-STF947 (MIXER/EJECTOR)



LRC005-A28

FIGURE 4-6. GENERAL ENGINE FLOW LAYOUT AND SUPPRESSOR HARDWARE - P&W-TBE-STJ950

- P&W STJ950 single-spool nonaugmented turbine bypass engine (TBE) with a convergent-divergent ejector nozzle with chute suppressor (see Figure 4-6 for engine flow arrangement).

Two important parameters in engine performance evaluations are cruise overall efficiency and takeoff thrust-to-weight ratio. Cruise overall efficiency, which is a measure of the efficiency of fuel energy conversion to jet kinetic energy, is very important because approximately 35 to 40 percent of the take-off gross weight is fuel burned during cruise. Another important engine performance parameter is specific fuel consumption during climb, which accounts for approximately 25 percent of the total block fuel. Takeoff thrust-to-weight ratio is important because engine thrust requirements are generally sized by takeoff field length. This will affect achievement of FAR Part 36, Stage 3, sideline and takeoff noise limits.

Cruise overall efficiencies for engines along with comparisons with the data from studies reported in Phase III (Reference 4-1) are summarized in Figure 4-7. Consistent with previous trends, the TBE has the slightly better cruise overall efficiency inherent with turbojets. The various current turbofans have overall efficiencies that are below the Phase III trends. This is due primarily to the engine companies' choosing a less aggressive technology standard associated with TAD of 1995 rather than 2000 for the Phase IIIA engines. This caused a 100°F reduction in design compressor discharge temperature (T_3) and a corresponding reduction in engine overall pressure ratio (OPR). The engine overall efficiency data are also compared with the potential value, which indicates the need for further development in engine and component technology.

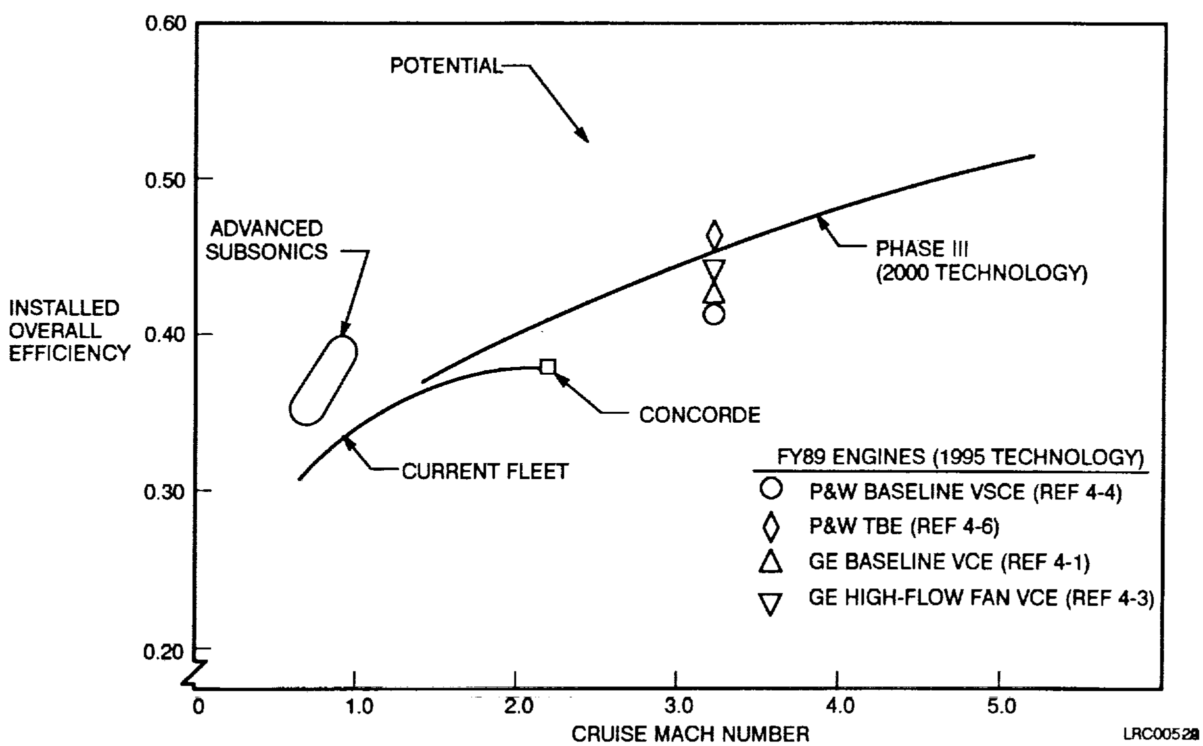


FIGURE 4-7. HSCT ENGINE OVERALL EFFICIENCY AT CRUISE

The engine thrust-to-weight ratio for the various engines along with a comparison with corresponding Phase III results are summarized in Figure 4-8. Some trends can be noted:

- The high-flow engine/nozzle concepts show a decided advantage over the high-specific-thrust engine plus suppressor concepts at the same bulk average jet velocity.

- The Phase IIIA GE Study A2 high-flow fan VCE with a suppressor has a significant thrust/weight advantage over both the Study A1 and Study M1 baseline VCEs.
- The TBE tends to have the lowest thrust-to-weight ratio of those engines studied.

In Figure 4-8, the choice of a reference engine exhaust jet velocity of 2,000 fps was arbitrary. However, it does show the relative performance of the engines at a partial power associated with lower jet velocity and hence lower noise.

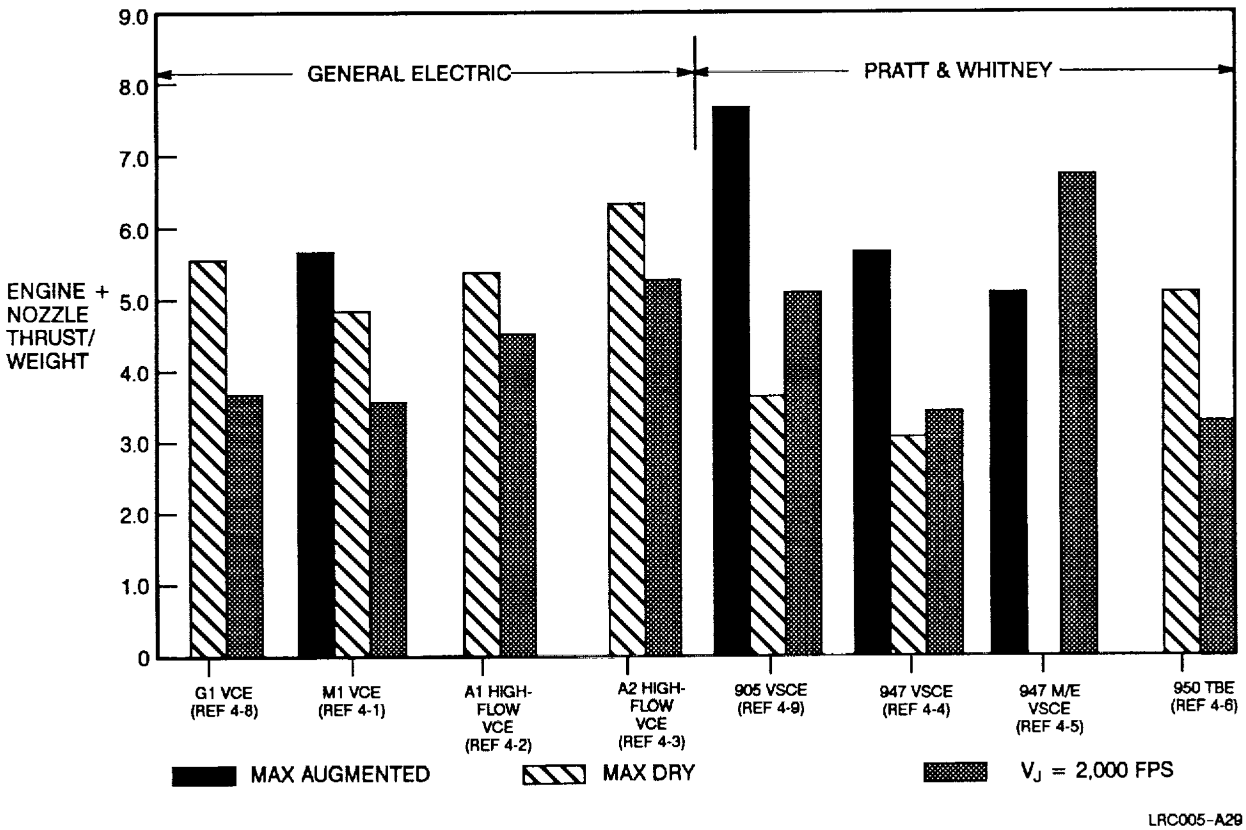


FIGURE 4-8. MACH 3.2 ENGINE SLS THRUST/WEIGHT

4.2.1 GE Engine Data Summary

All three GE engines have essentially the same gas generator, which is similar to that on the GE21/F14, Study G1, VCE evaluated in the Phase III studies. The VCE is a double-bypass, variable-cycle engine that employs variable-geometry features to vary the bypass ratio to optimize the compressor-turbine match over the entire flight profile and to permit high-flowing the engine, i.e., maintaining the maximum corrected airflow for a range of power settings to minimize inlet spillage and bypass drags at cruise.

In contrast to the nonaugmented Study G1, VCE, the Study M1 engine incorporates a low ΔT augmentor to provide additional thrust during takeoff, transonic acceleration, and climb. The Study M1 VCE incorporates features to obtain an inverted velocity profile at takeoff by routing the bypass air inward through struts and discharging it through a chute suppressor in the aft centerbody. The resulting jet velocities as a function of takeoff thrust are shown in Figure 4-9.

The Study A1 and Study A2 high-flow fan VCEs have the same gas generator (but no augmentor) as the Study M1 VCE, but with the added feature of a high-flow fan stage, which increases the total

engine airflow at takeoff by approximately 50 percent with a 20-percent increase in thrust, but with an associated weight penalty. In the Study A1 engine at takeoff, the fan secondary flow is routed through struts and discharged through chutes in the aft centerbody, resulting in a two-stream configuration. Thus, the Study A1 concept retains the inverted velocity profile, and no suppressor has been provided. The resulting jet velocities as a function of takeoff thrust are shown in Figure 4-10. During cruise, the inlet to the high-flow fan stage is closed off except for a small airflow for fan cooling, and thus the engine operates as a low-bypass turbofan.

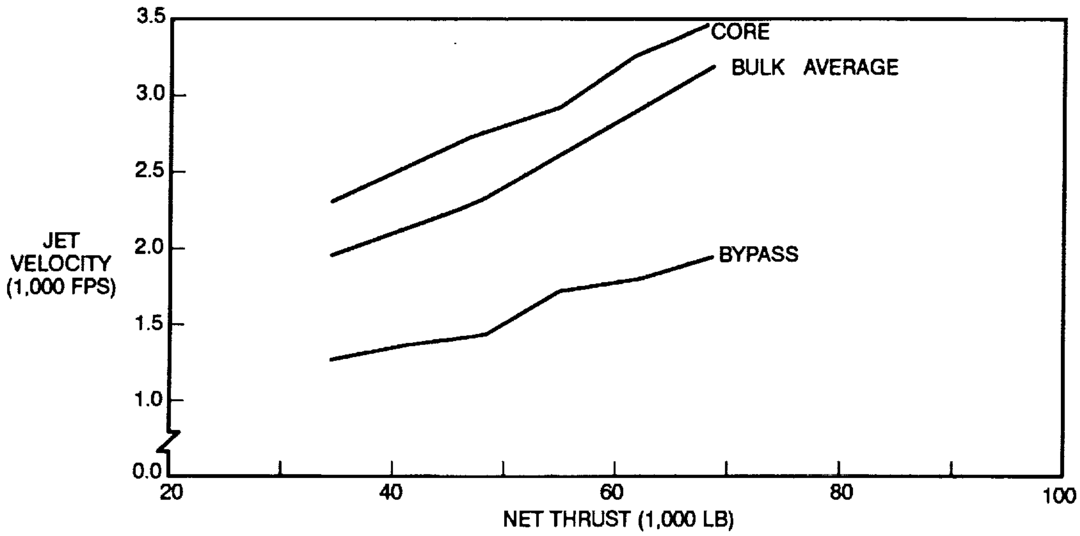


FIGURE 4-9. JET VELOCITY VERSUS NET THRUST - GE 21/F14 VCE STUDY M1

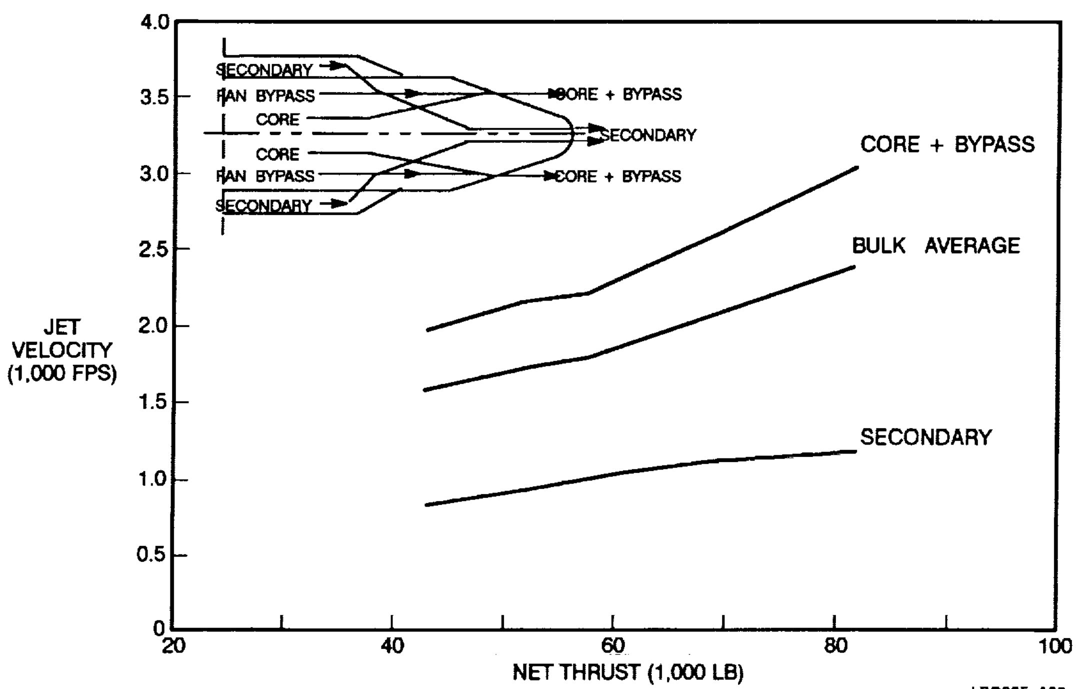


FIGURE 4-10. JET VELOCITY VERSUS NET THRUST - GE21/FLA1-STUDY A1

In the Study A2 concept at takeoff, the high-flow fan discharge is routed through its own short cowl duct and nozzle, and does not mix internally with the other engine streams, resulting in a three-stream engine. The bypass stream is routed through struts in the same manner as on the Study M1 VCE, and thus there is an inverted velocity profile in the inner two streams surrounded by a low-velocity secondary stream at the exhaust exit plane. The resulting jet velocities as a function of takeoff thrust are shown in Figure 4-11. Preliminary noise estimates did not include the effects of a noise suppressor but show that one is required. The weights and performance for this engine did include a suppressor for sizing purposes.

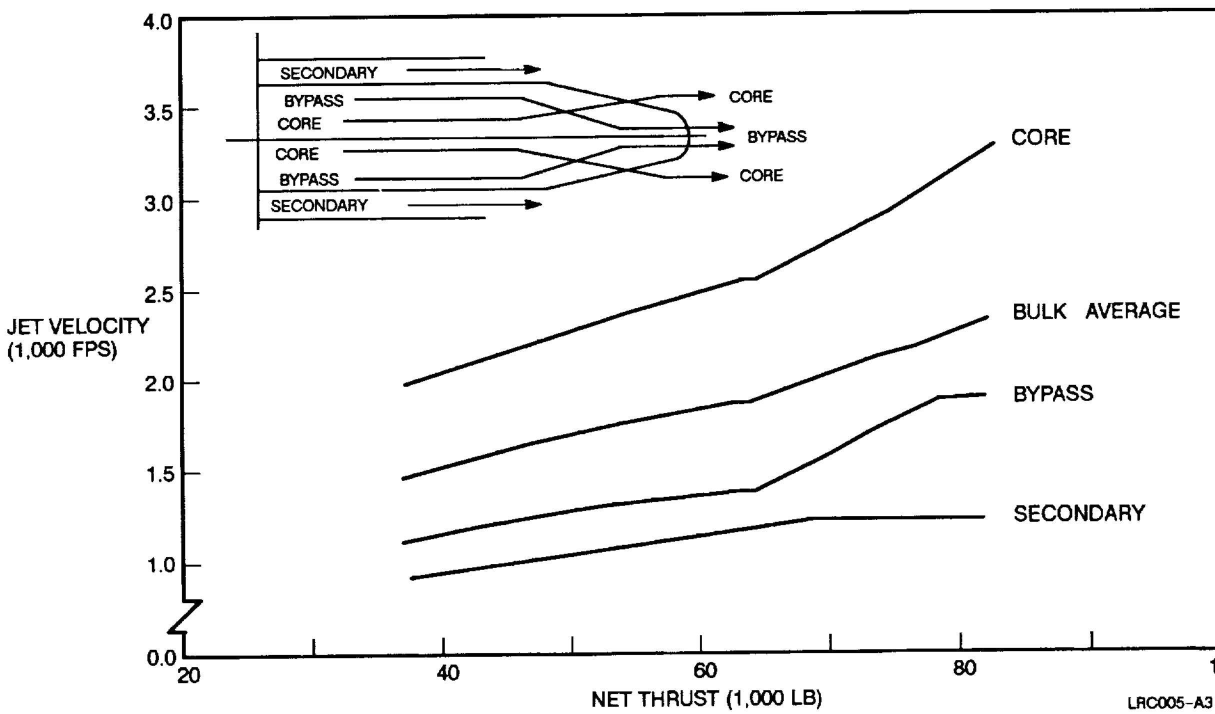


FIGURE 4-11. JET VELOCITY VERSUS NET THRUST - GE21/FLA1-STUDY A2

4.2.2 P&W Engine Data Summary

The STF947 VSCE is an advanced, moderate-bypass, nonmixed-flow turbofan with duct burner augmentation and a coannular nozzle to produce an inverted velocity profile at takeoff for jet noise reduction. This VSCE cycle was an updated version of the STF905 VSCE. A distinctive operating feature is the independent control of both the core and fan bypass stream temperatures and velocities for in-flight cycle matching. In addition to providing the inverted velocity profile at takeoff, this independent control of the two streams permits high-flowing the engine. The VSCE jet velocities as function of takeoff thrust are shown in Figure 4-12. The VSCE also incorporates a duct stream tube/chute suppressor, which reduces the basic takeoff thrust by approximately 4 percent, based on a Phase III study.

The gas generator for the VSCE with the high-flow mixer/ejector nozzle is essentially the same as for the baseline VSCE except for a small increase in duct burner size. Instead of employing an inverted velocity profile to reduce noise, the mixer/ejector nozzle concept incorporates additional components designed to increase the total engine mass flow by approximately 100 percent. This destroys the inverted velocity profile and results in a small increase in weight and overall engine diameter. A representative takeoff jet velocity point for the mixer/ejector concept is shown in Figure 4-12.

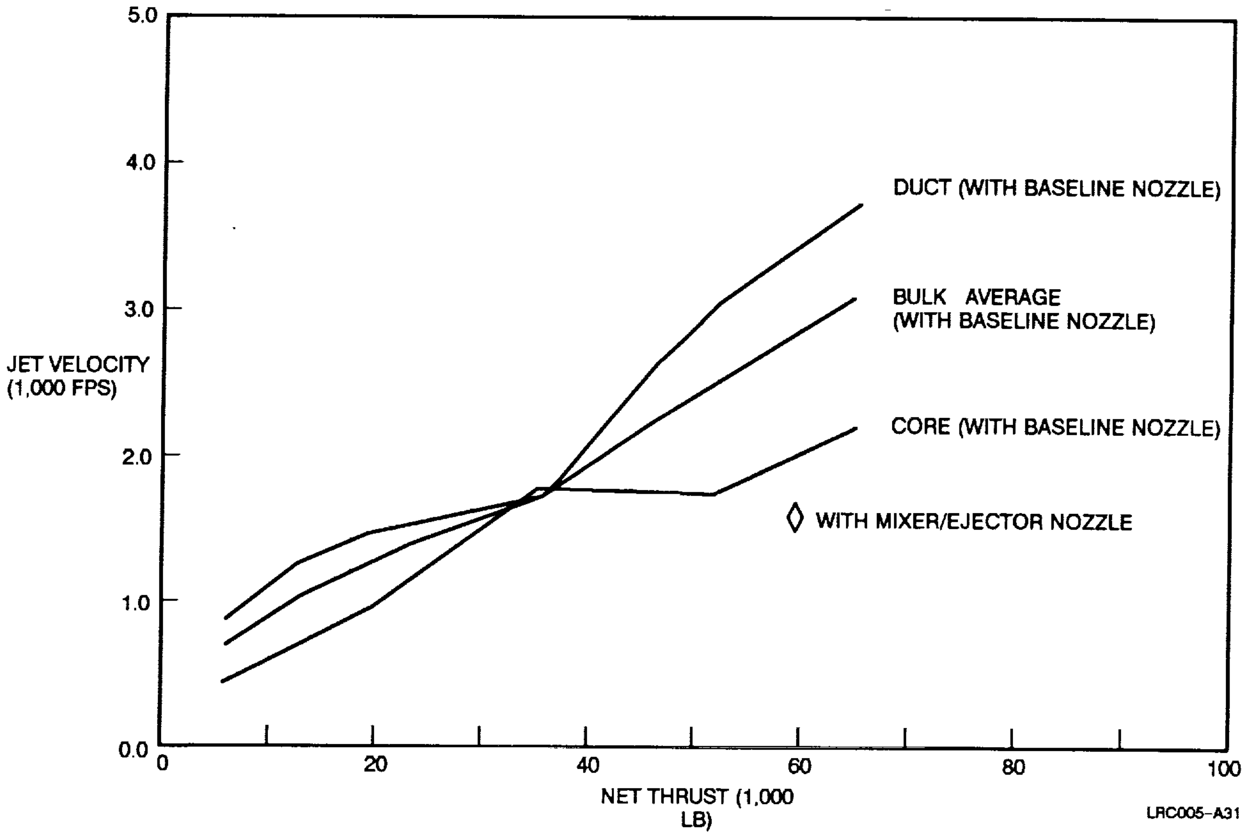


FIGURE 4-12. JET VELOCITY VERSUS NET THRUST - P&W STF947 VSCE

The STJ950 TBE is a single-spool, nonaugmented turbine bypass engine. At takeoff and other high-power operating modes, a portion of the compressor discharge air is bypassed around the combustor and turbine and mixed with the turbine exhaust stream in the nozzle. At cruise, in order to maximize efficiency, all of the compressor discharge air goes through the combustor and turbine, and hence the engine operates as a conventional turbojet with better specific fuel consumption than for a turbofan. However, the engine requires a noise suppressor because of the inherent high jet velocities at takeoff (Figure 4-13), and all analyses to date have assumed a convergent-divergent ejector nozzle with chute suppressor. This engine can also be provided with the mixer/ejector nozzle, and future studies will evaluate this concept.

4.3 NOISE SUPPRESSION ASSUMPTIONS

In order to conduct a noise screening evaluation of the acoustic engine data bases given in Section 4.2, it was necessary to determine a noise suppression curve as a function of bulk average jet velocity (Figure 4-14). This was determined from discussions held with the engine companies for a certification date of approximately 2005 (TAD = 1995). Where appropriate, an inverted velocity profile (IVP), noise suppressor, and accoustically lined ejector were employed. Noise attenuation as a function of bulk average exhaust jet velocity is given in Figure 4-14 for the IVP effect alone, a total attenuation for the combination of IVP, duct stream suppressor and lined ejector, and a suppressor/ejector configuration for a single-flow-stream configuration.

At high jet velocities (e.g., 2,800 fps) for the above suppressor nozzle concepts, it may be possible to increase the noise suppression by about 3 EPNdB for a certification date of 2010 (TAD = 2000).

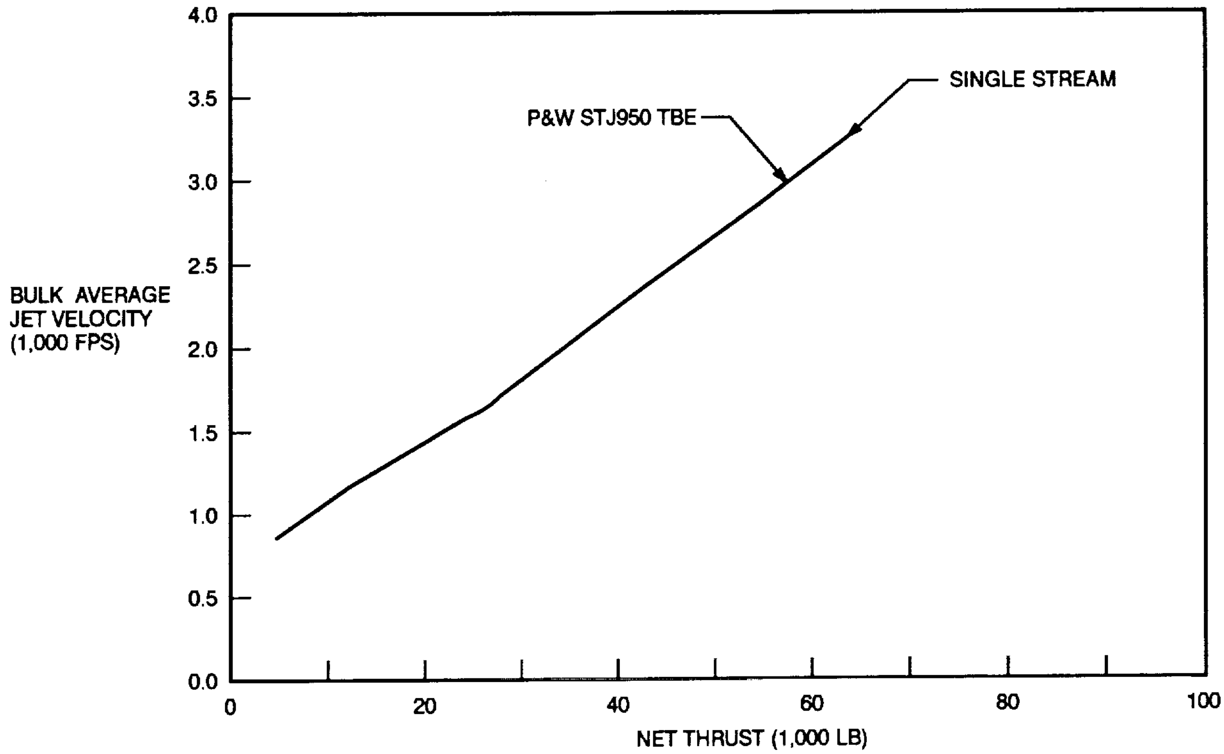


FIGURE 4-13. JET VELOCITY VERSUS NET THRUST – P&W STJ950 TBE

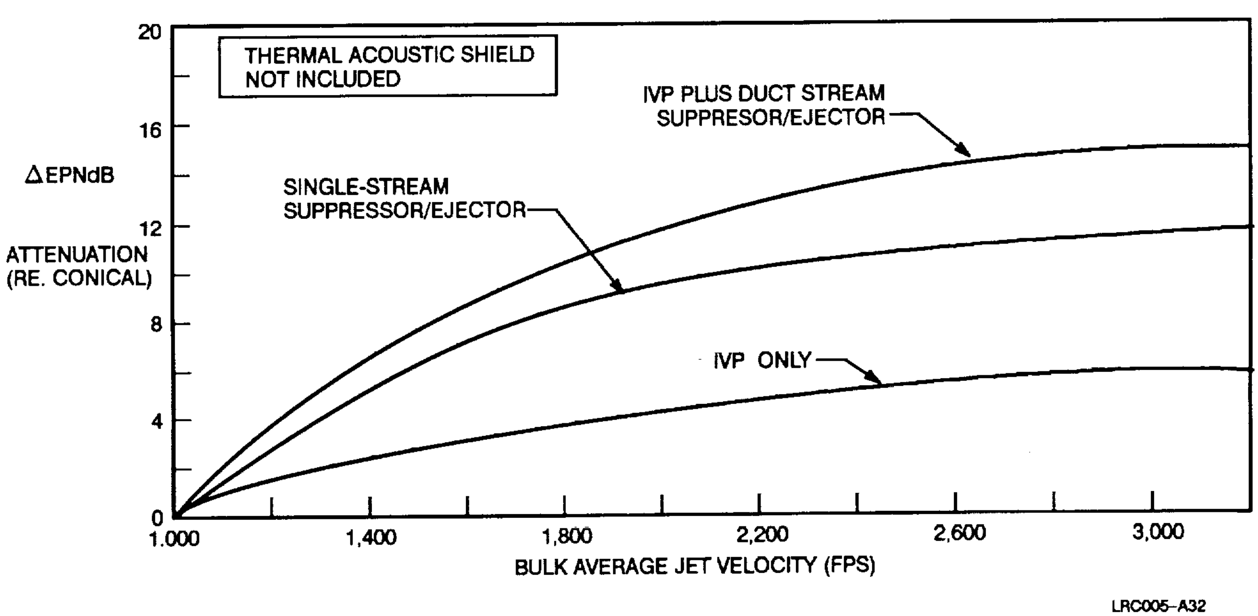


FIGURE 4-14. NOISE SUPPRESSOR ASSUMPTIONS

The engine cycles described in Section 4.2 used combinations of the above noise suppression devices. A summary of the noise suppression hardware employed for each configuration is given in Table 4-1.

TABLE 4-1
NOISE SUPPRESSION HARDWARE

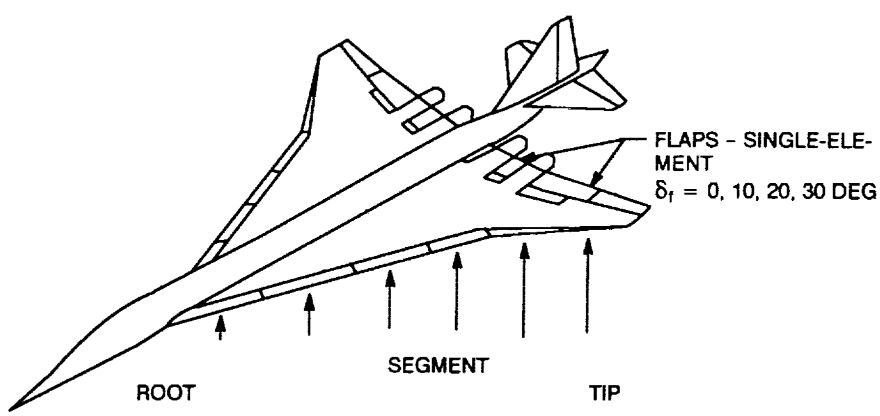
ENGINE CONCEPT	INVERTED VELOCITY PROFILE	DUCT STREAM SUPPRESSOR	SINGLE-STREAM SUPPRESSOR	MIXER/ EJECTOR NOZZLE	ACOUSTICALLLY LINED EJECTOR
GE STUDY M1 VCE	YES	YES	NO	NO	YES
GE STUDY A1 VCE	YES	NO	NO	NO	NO
GE STUDY A2 VCE	YES	NO*	NO	NO	NO
P&W TBE	NO	NO	YES	NO	YES
P&W VSCE	YES	YES	NO	NO	YES
P&W VSCE (MIXER/EJECTOR)	NO	NO	NO	YES	YES

*SUPPRESSOR WEIGHT WAS INCLUDED FOR AIRCRAFT SIZING PURPOSES

LRC005-A38

4.4 EFFECTS OF AIRCRAFT SIZING AND PERFORMANCE ON NOISE

The D3.2-3A configuration (see appendix) was used for the aircraft sizing studies. The low-speed aerodynamics assumed leading edge flap deflection with Schedule R (see Figure 4-15) using 30-degree partial-span flaps with 80-percent leading edge suction on takeoff and 10-degree full-span flaps with 80-percent leading edge suction on approach. The low-speed polars for various flap settings of the D3.2-3A concept are shown in Figure 4-16 with the 80-percent suction polar, which is considered an achievable goal. The sizing studies were conducted using flight performance assumptions based on



SCHEDULE	1	2	3	4	5	6
CLEAN	0	0	0	0	0	0
R	13	34	35	35	19	29

LEADING EDGE FLAP DEFLECTION SCHEDULES

LRC005-A33

FIGURE 4-15. HIGH-LIFT SYSTEM FOR D3.2-3A

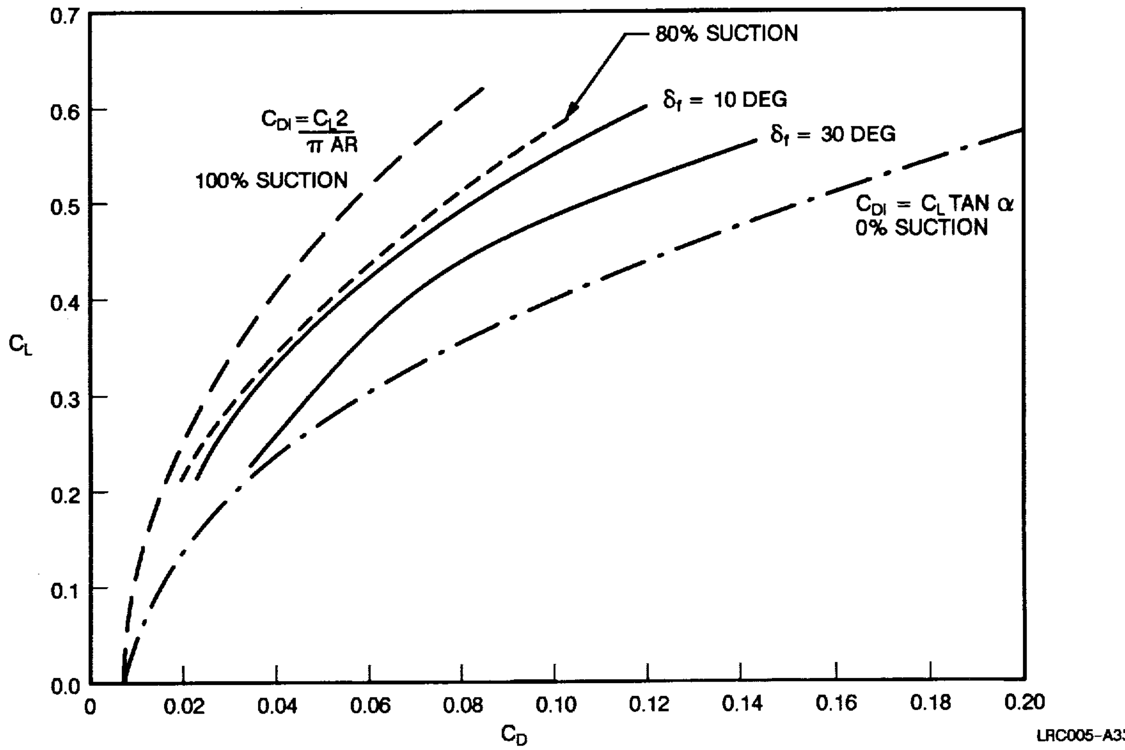


FIGURE 4-16 TRIMMED LOW-SPEED POLARS FOR D3.2-3A

commercial domestic and international rules and practices. The analysis included takeoff and landing performance, mission analyses, and determination of takeoff and approach flight paths.

4.4.1 Mission Definition

The mission profile used for the aircraft sizing studies is shown in Figure 4-17 and begins with conventional takeoff and climb to 10,000 feet. This is followed by an accelerating climb to the cruise Mach number. The climb is continued at cruise Mach number until optimum cruise altitude is reached. The profile is the same as the one used in the Phase III studies. The main limitations in climb are the cabin rate of climb and the aircraft excess thrust over drag at the top of climb. Conventional cabin pressure altitude at cruise is 8,000 feet, and the limiting rate of pressurization change is equivalent to 300 feet per minute at sea level. This requires the climb to take at least 23.5 minutes. A requirement of 4,000-feet-per-minute potential rate of climb was assumed to ensure sufficient acceleration and rate of climb to reach cruise altitude.

Cruise is flown at constant Mach number and optimum altitude to maximize range factor, which is mainly a function of the maximum lift-to-drag ratio and engine specific fuel consumption. During fuel burn-off, the aircraft is allowed to cruise-climb to remain at optimum conditions. The descent is at idle power and constant airspeed, as is the current convention. The cabin rate of descent is limited to 300 feet per minute with idle power.

Below 10,000 feet in altitude, regulation-specified speeds of 250 keas are maintained until landing approach at a conventional speed of 140 to 160 knots. Fuel reserves based on international rules are maintained: 5 percent of block fuel, fuel to fly to an alternative destination of 200 nautical miles, and other appropriate fuel determinations.

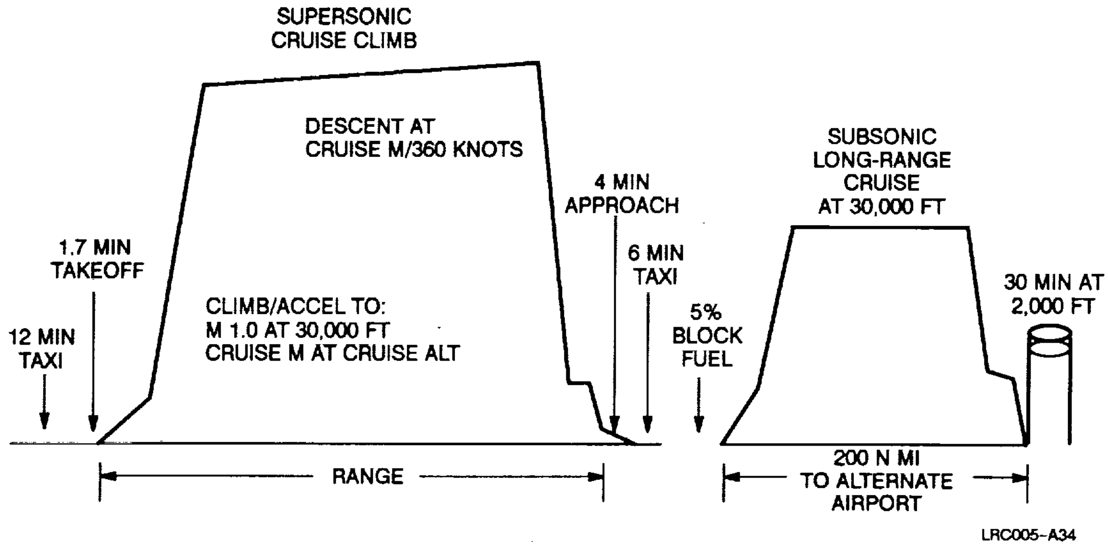


FIGURE 4-17. MISSION PROFILE

4.4.2 Aircraft Sizing

Sizing was accomplished using the Computer-Aided Sizing and Evaluation System (CASES), which consists of interacting modules of aerodynamics, stability and control, propulsion, weight and balance, systems, and airplane performance. The weight analysis included the use of conceptual design weight methodology, advanced aluminum metal matrix composite structures, and advanced aircraft system concepts consistent with the Phase III D3.2-3A baseline concept.

The combined structure and systems weight included the wing, fuselage, thermal protection system, horizontal tail, vertical tail, landing gear, fuel system, flight guidance and controls, furnishings, environmental control system, instruments, electrical power and lighting, avionics, ice protection, aircraft ground handling, a partial wing laminar flow control system, and operational items.

The power plant system weight included the variable-geometry bicone inlet structure and system, nacelle, functional engine systems, mounting structure, bare engine, nozzle, acoustic suppressor when applicable, and a thrust reverser system.

Parametric weights for the six engine concepts were generated using individually tailored inlet geometries developed by Douglas and engine weights provided by the engine manufacturers (engine weight includes bare engine, exhaust nozzle, acoustic suppressor when applicable, and thrust reverser). The engine weights reflected a 1995 engine technology availability date.

By varying maximum takeoff weight, the aircraft range was determined, taking into account constraints and margins as considered appropriate. The constraints were (1) takeoff field length of 10,600 feet for an ISA day, (2) landing approach speed of approximately 140 knots, and (3) cruise at optimum altitude or at the operationally determined ceiling (4,000-feet-per-minute potential rate of climb).

4.4.3 Sizing and Performance Results

The results of the aircraft sizing and performance are given in Figures 4-18 and 4-19, which show sea level static thrust and TOGW versus aircraft range, respectively. The effects of aircraft sizing on sideline noise is shown in Figure 4-20 as a function of aircraft range. Each configuration was assessed at the highest power code or setting (maximum rated thrust available from the engine). The maximum range achieved by each configuration is summarized in Table 4-2 for selected takeoff gross weights.

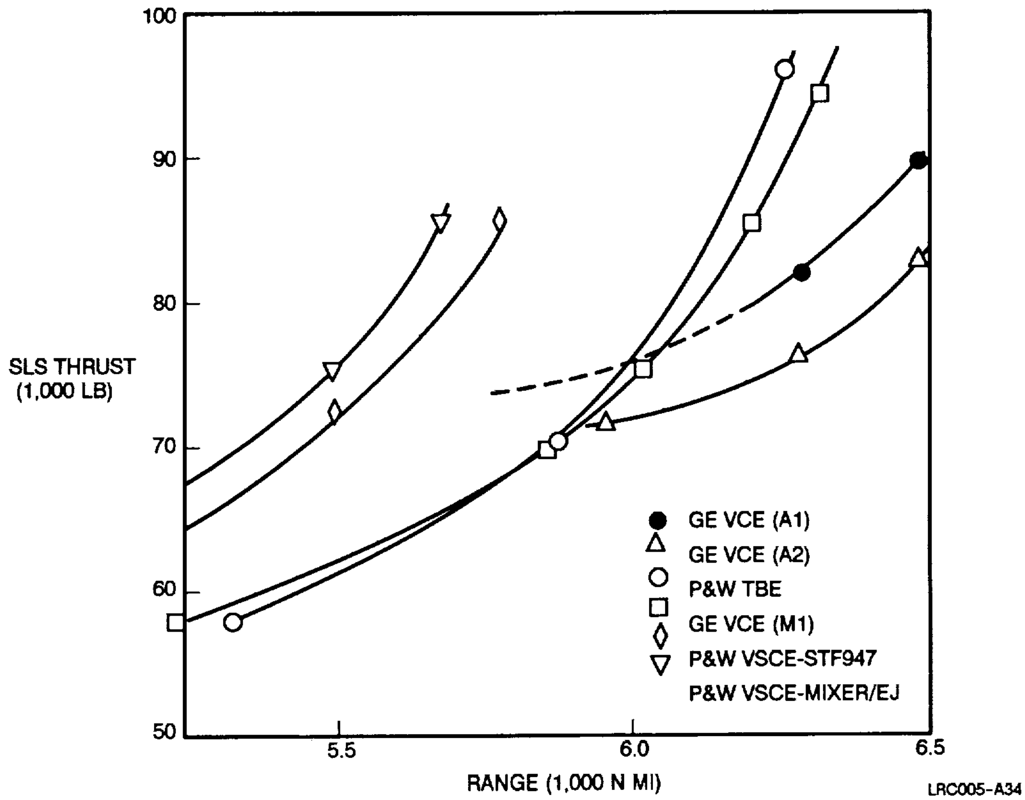


FIGURE 4-18. AIRCRAFT SIZING - SLS THRUST VERSUS RANGE

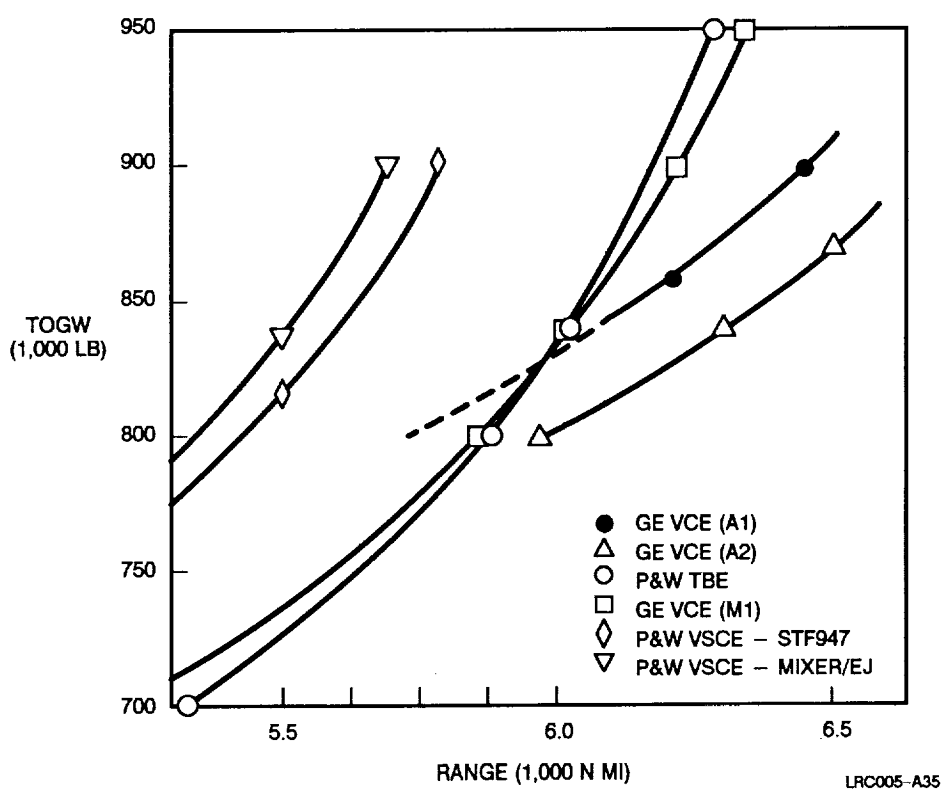


FIGURE 4-19. AIRCRAFT SIZING – TOGW VERSUS RANGE

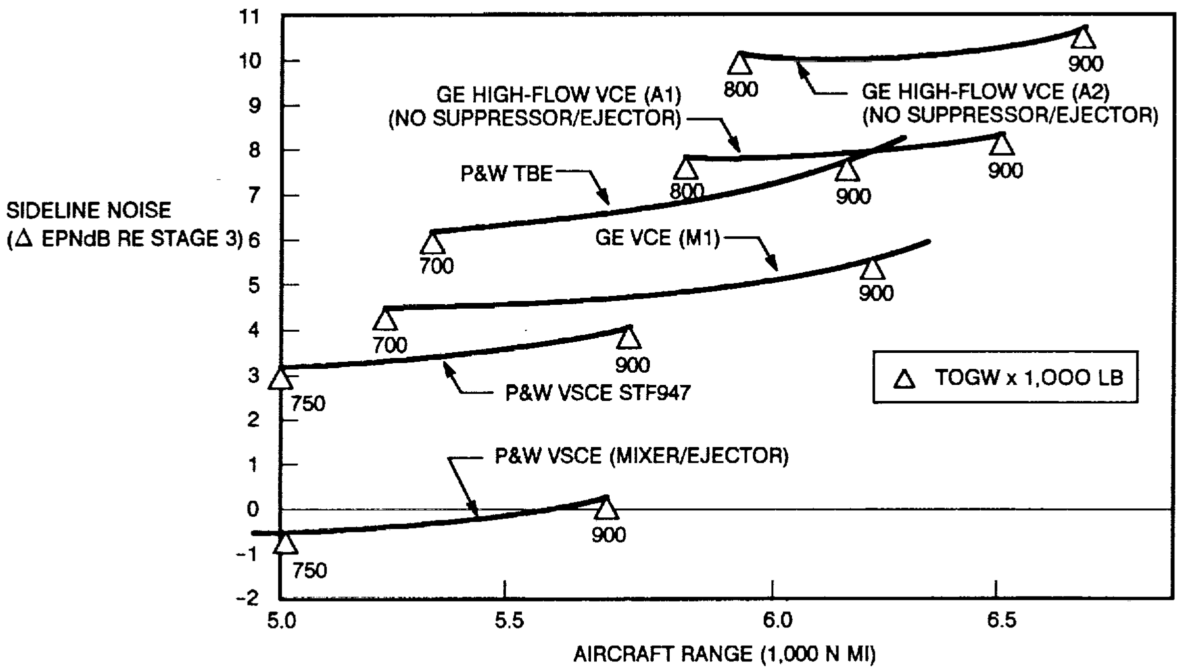


FIGURE 4-20. EFFECTS OF AIRCRAFT SIZING ON SIDELINE NOISE AND RANGE

TABLE 4-2
D3.2-3A RESULTS OF AIRCRAFT SIZING FOR NOISE

ENGINE TYPE	POWER CODE OR SETTING	TAKEOFF V_J (FPS)	SLS ISA (1,000 LB)	MTOW (1,000 LB)	AIRCRAFT RANGE (N MI)	SIDELINE NOISE (EPNdB)	EPNdB RE STAGE 3
P&W TBE	50	3,210	77.3	850.0	6,180	110.0	+7.1
P&W VSCE	100	3,180	85.5	900.0	5,790	107.0	+4.0
P&W VSCE (MIXER/EJECTOR)	100	1,500	85.5	900.0	5,690	103.2	+0.2
GE STUDY M1 VCE	100	3,210	75.5	841.0	6,030	107.9	+5.1
GE STUDY A1 VCE	50	2,410	89.5	913.0	6,500	111.3	+8.5
GE STUDY A2 VCE	50	3,000	82.3	871.0	6,500	113.5	+10.5

NOTE: AIRCRAFT SIZING CONDUCTED AT THE MAXIMUM RATED THRUST CONDITION FOR EACH DATA BASE LRC005-A39

The sized weight and geometry analysis produced for the aerodynamic analysis is summarized in Table 4-3 for selected conditions.

The following sizing results have been selected for comparison at an MTOW of 800,000 pounds:

GE Results

- The Study M1 VCE engine is estimated to be approximately 5 EPNdB above the Stage 3 noise limit on sideline for a takeoff maximum-rated sea level thrust (SLS) of 70,000 pounds. A range of 5,850 nautical miles was achieved.

TABLE 4-3
WEIGHT BREAKDOWN

ENGINE TYPE	STRUCTURE/ SYSTEMS (LB)	POWER PLANT SYSTEMS (LB)	OPERATING EMPTY WEIGHT (LB)	PAYLOAD (LB)	FUEL WEIGHT (LB)	MTOW (LB)
P&W TBE	206,300	72,550	278,850	61,500	509,650	850,000
P&W VSCE	209,400	74,350	283,750	61,500	554,750	900,000
P&W VSCE (MIXER/EJECTOR)	209,400	80,350	289,750	61,500	548,750	900,000
GE STUDY M1 VCE	205,750	68,650	274,400	61,500	505,100	841,000
GE STUDY A1 VCE	210,200	82,000	292,200	61,500	559,300	913,000
GE STUDY A2 VCE	207,600	66,000	273,600	61,500	535,900	871,000

LRC005-A40

- The Study A1 VCE (two-stream exhaust) is estimated to be approximately 8 EPNdB above the Stage 3 noise limit on sideline for a takeoff at maximum-rated SLS thrust of 74,000 pounds. A range of 5,800 nautical miles was achieved. A noise suppressor must be integrated into the exhaust system.
- The Study A2 VCE (three-stream exhaust) is not the best approach to achieve low bulk average jet velocities (i.e., approximately 10 EPNdB above the Stage 3 noise limit on sideline). A maximum-rated SLS takeoff thrust of 71,000 pounds was required and a range of 5,950 nautical miles was achieved.
- For a given TOGW, the Study A2 VCE achieved an increased range of approximately 150 nautical miles compared with the Study A1 VCE. However, the Study A1 VCE achieved 2-3 EPNdB lower jet noise level.
- Both the A1 and A2 study engines achieved 6,500 nautical miles at an MTOW of 900,000 pounds with an associated increase of SLS thrust of approximately 15,000 pounds and an increase of approximately 0.5 to 1.0 EPNdB in sideline noise.

P&W Results

- The STJ950 TBE is estimated to be approximately 6 to 7 EPNdB above the Stage 3 noise limit on sideline for a takeoff at a maximum-rated SLS thrust of 68,000 pounds. A range of 5,800 nautical miles was achieved.
- The STF947 VSCE is estimated to be 3 to 4 EPNdB above the Stage 3 noise limit on sideline for a takeoff at a maximum-rated SLS thrust of 69,000 pounds. A range of 5,400 nautical miles was achieved.
- Preliminary sizing results using the mixer/ejector nozzle for the VSCE STF947 indicate that at maximum-rated conditions the TOGW and range are essentially the same as for the baseline nozzle (suppressed) but at a lower noise level of approximately 3 to 4 EPNdB.

4.5 ACOUSTIC TECHNOLOGY SCREENING

Noise certification acoustic technology was screened for the engine data bases given in Section 4.2. Sideline, takeoff and approach noise estimates were determined for an MTOW of 800,000 pounds and an MLW of 420,000 pounds. For the P&W-VSCE (mixer/ejector) concept engine, data were available for sideline only. For each engine concept, jet noise (mixing plus shock) estimates were calculated and, in the case of the VSCE basic and mixer/ejector configurations, duct burner noise in the suppressed mode was estimated and added to jet noise. Other noise sources such as turbomachinery,

core (main combustor), and airframe noise were not predicted at this time. These additional noise sources will be evaluated in future studies.

To evaluate engine acoustic technology efficiency (e.g., engine cycle and noise suppression devices) realistic aircraft/engine parameters and assumptions were selected from previous mission performance and sizing analyses (see Table 4-4).

The results of the acoustic technology noise screening estimates relative to the Stage 3 noise limits are given in Table 4-5. The noise estimates for each configuration are given in Figures 4-21, 4-22, and 4-23 for sideline, takeoff, and approach, respectively.

The sideline noise screening results (Figure 4-21) show that engine cycles with noise-suppression devices providing large noise reductions (i.e., the P&W VSCE and GE VCE) are 4 to 5 EPNdB above the Stage 3 noise limit. However, the P&W VSCE with the mixer/ejector nozzle was estimated to be 1 EPNdB below the Stage 3 limit. This ocurred because the mixer/ejector system approximately halved the exhaust jet velocity compared with the above the P&W VSCE configuration.

TABLE 4-4
AIRCRAFT/ENGINE ACOUSTIC TECHNOLOGY SCREENING ASSUMPTIONS

(ISA + 10°C)			
	SIDELINE	TAKEOFF	APPROACH
MTOW/MLW (LB)	800,000	800,000	420,000
NET THRUST AT 1,000 FT (LB)	59,000	–	–
NET THRUST AT 1,200 FT (LB)	–	40,000	–
NET THRUST AT 400 FT (LB)	–	–	10,000
AIRCRAFT SPEED (KNOTS) TAS	201	201	168

LRC005-A41

TABLE 4-5
SUMMARY OF ACOUSTIC TECHNOLOGY SCREENING RESULTS

JET NOISE Δ EPNdB RE. STAGE 3 LIMIT			
ENGINE	SIDELINE	TAKEOFF	APPROACH
P&W TBE	+6 (-12)	+6.5 (-11)	-7.5 (0)
P&W VSCE	+3* (-14)	3* (-12)	-3.5 (0)
P&W VSCE (MIXER/EJECTOR)	-1 (0)	–	–
GE STUDY M1 VCE	+4 (-14)	+2 (-12)	-3 (0)
GE STUDY A1 VCE	+7 (-3)	+3 (-2.5)	-9.5 (0)
GE STUDY A2 VCE	+9 (-4.5)	+6 (-3.5)	-6 (0)

* INCLUDING DUCT BURNER NOISE CONTRIBUTION

( ) NOISE SUPPRESSION INCREMENT OTHER THAN ENGINE CYCLE

LRC005-A42

The takeoff noise screening was conducted at a 1,200-foot altitude and at a cutback net thrust of 40,000 pounds. The P&W VSCE and GE VCE are estimated to be 2 to 3 EPNdB above the Stage 3 noise limit (see Figure 4-22). However, using takeoff flight profiles that assume optimized takeoff performance, a takeoff altitude of 1,500 feet, and a cutback thrust level of 38,000 pounds, the noise level was estimated to be below the Stage 3 noise limit.

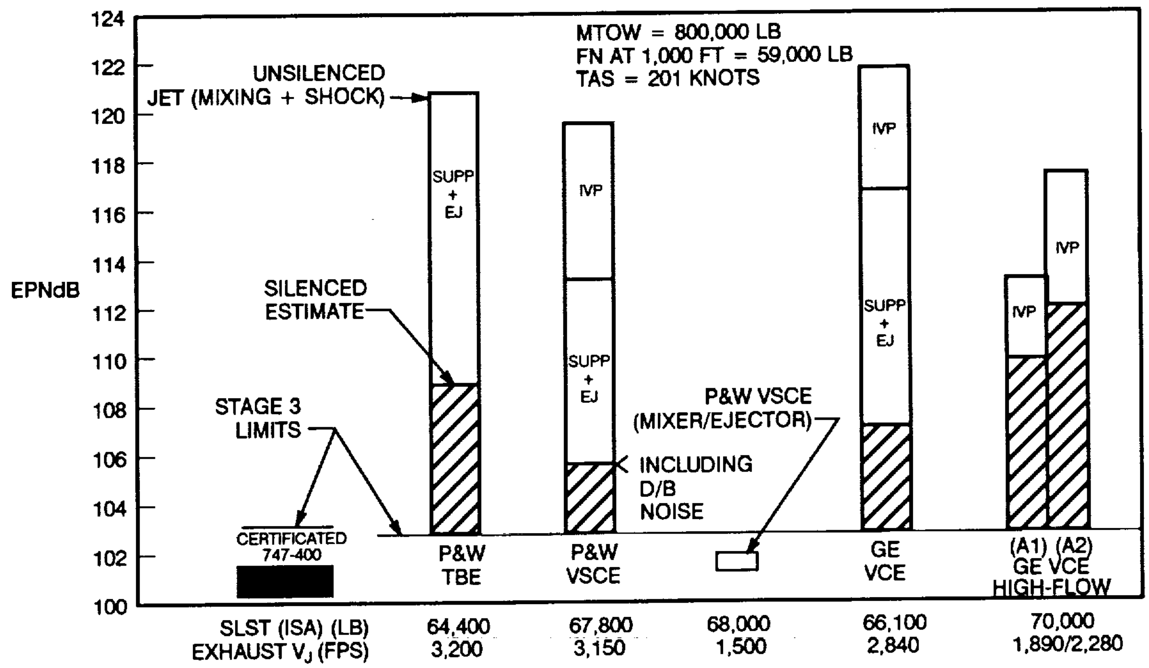


FIGURE 4-21. ACOUSTIC TECHNOLOGY RESULTS - SIDELINE NOISE

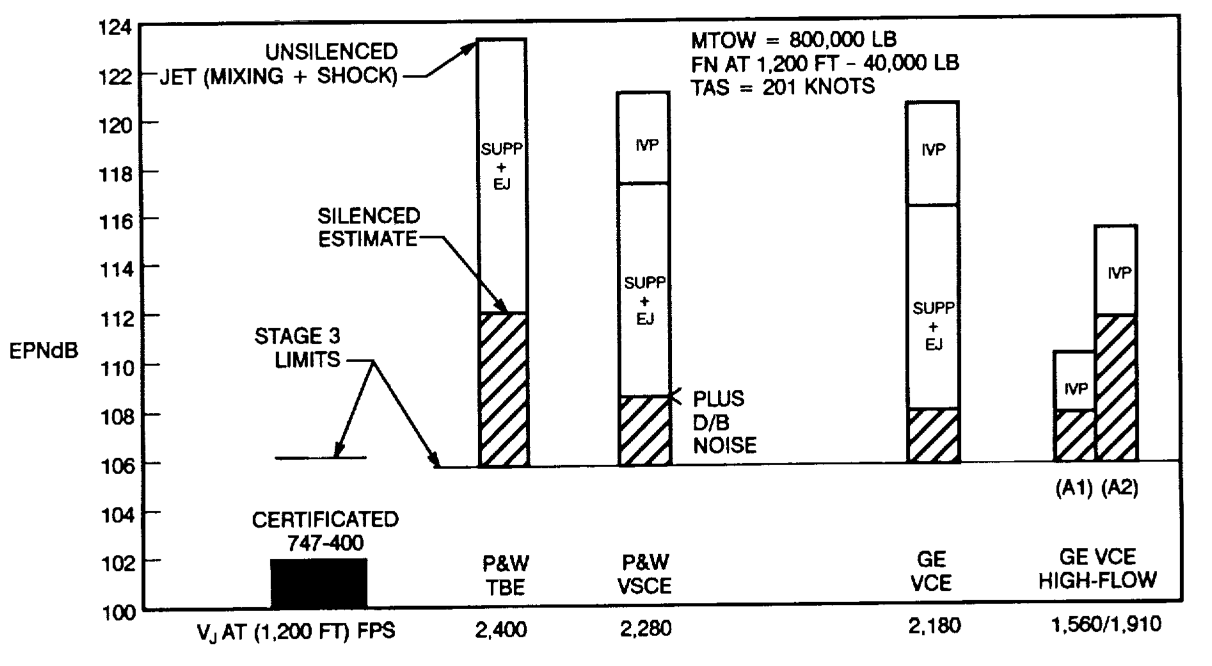


FIGURE 4-22. ACOUSTIC TECHNOLOGY RESULTS - TAKEOFF NOISE

Approach jet noise estimates (Figure 4-23) for all engine cycles screened are below the Stage 3 noise limit. Since these estimates include only jet noise (mixing and shock), a more complete assessment of other noise sources (e.g., turbomachinery, airframe, and combustion noise) should be conducted to determine their significance.

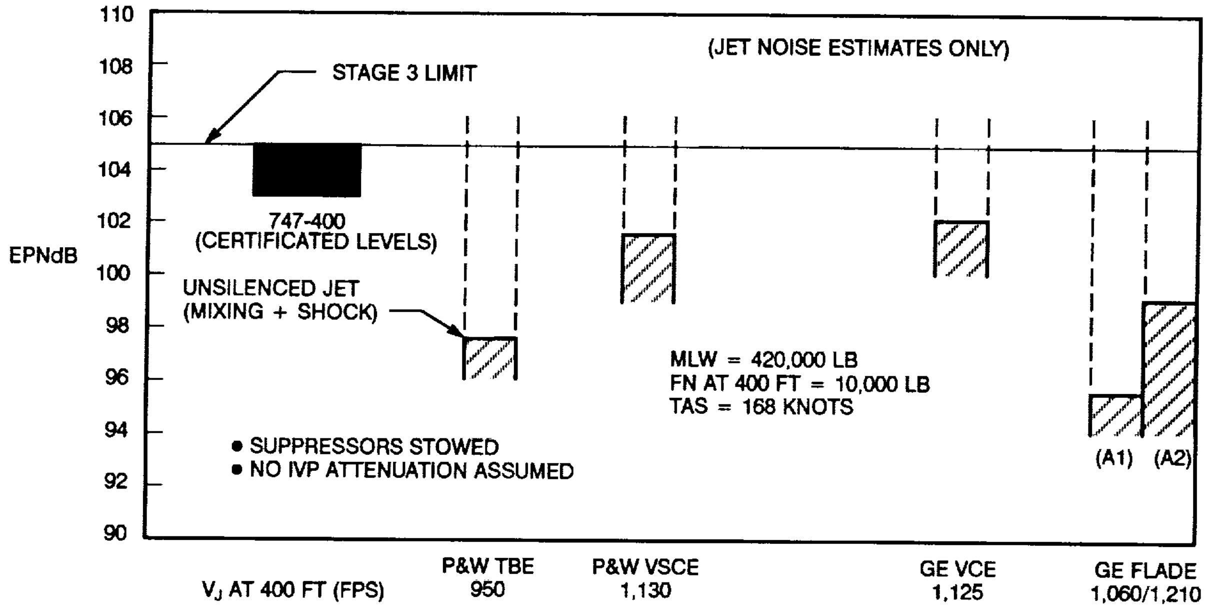


FIGURE 4-23. ACOUSTIC TECHNOLOGY RESULTS – APPROACH NOISE

4.6 AIRPORT NOISE

It is important to evaluate the noise impact at international airports of adding HSCTs to the world fleet. Figures 4-21, 4-22, and 4-23 compare the predicted HSCT noise certification levels derived from the acoustic technology screening studies with the Boeing 747-400 levels for sideline, takeoff, and approach, respectively. These figures show that the 747-400 is well below current Stage 3 noise limits at sideline and takeoff and close to the limit on approach. Therefore, for the HSCT to attain airport noise levels compatible to those of the 747-400, it may be necessary to develop automated minimum noise abatement procedures during takeoff and approach.

4.7 CONCLUSIONS

The following general conclusions can be drawn from the engine performance and noise analyses of the six engine configurations evaluated for a constant MTOW of 800,000 pounds.

4.7.1 GE Engines

- The Phase IIIA GE Study A2 high-flow fan VCE with a suppressor has a significant thrust/ weight advantage over both the Study G1 and Study M1 baseline VCEs.

- The noise sizing results are summarized below for the VCE:

	Study M1	Study A1	Study A2
SLS (lb)	70,000	74,000	71,000
Range (n mi)	5,850	5,800	5,950
Sideline noise level re. Stage 3 (ΔEPNdB)	+5	+8	+10

- Both the A1 and A2 study engines achieved 6,500 nautical miles at an approximate MTOW of 900,000 pounds. However, the sideline noise increased by 0.5 to 1.0 EPNdB.

4.7.2 P&W Engines

- The tube/chute suppressor reduces baseline VSCE takeoff thrust by approximately 4 percent.
- Based upon preliminary P&W data, the mixer/ejector nozzle concept increases the exhaust mass flow by approximately 100 percent during takeoff.
- The noise sizing results are summarized below for the VSCE and TBE:

	TBE	VSCE
SLS (lb)	68,000	69,000
Range (n mi)	5,800	5,400
Noise Level re. Stage 3 (ΔEPNdB)	+6	+3

- The TOGW and range for the VSCE mixer/ejector nozzle are essentially the same as for the baseline nozzle but at a lower noise level of approximately 3 to 4 EPNdB.

4.8 RECOMENDATIONS

4.8.1 Engine Cycle and Noise Suppressor Development

The following engine cycle and noise suppressor developments are recommended for further noise evaluations:

- Further development of the P&W mixer/ejector concept as applied to both the TBE and VSCE engine cycles.
- Noise evaluation of the P&W tandem fan concept, if deemed viable by P&W.
- Further development of the GE Study A1 and A2 high-flow fan VCE concepts incorporating a noise suppressor into the exhaust nozzle.
- Cycle refinements to optimize the engine cycle parameters (e.g., OPR, T_3, T_4, FPR, BPR, etc.) against aircraft thrust and noise requirements.
- Investigation of the effects of engine technology availability date on engine thrust/weight, specific fuel consumption, and noise.
- An aircraft sizing study of the effects of using lower engine power settings on noise and range.
- Investigation of alternative engine and nozzle concepts that will reduce aircraft TOGW and satisfy FAR Part 36 Stage 3, noise limits.

4.8.2 Operational Procedures

Takeoff and approach operational procedures should be developed for noise certification and minimizing airport noise. These should include the following:

- Sideline noise minimization (including certification levels) during takeoff using an automated power reduction technique.
- Engine sizing at brake release to improve the aircraft height at the takeoff monitor in conjunction with sideline noise reduction procedures.
- Takeoff noise minimization under the flight path involving deep engine cutback power procedures, aerodynamic configurational changes, and possible engine cycle changes.
- Cycle and engine power management procedures for aircraft climb to cruise altitude. The method of alleviating jet noise when stowing suppression devices needs careful consideration regarding noise impact.
- Minimizing approach noise through speed and glide slope angle.

4.8.3 Public Awareness Program

It will be necessary to involve the public on the progress of designing an environmentally acceptable HSCT. In order for the HSCT to operate worldwide, various environmental agencies or groups may be involved. The assistance of the International Civil Aviation Organization (ICAO) or some form of international body may be required to coordinate environmental assessments.

Overall, the public program should address, in addition to airport noise impact, the environmental issues of acceptable sonic boom levels overland and the effects of global engine emissions.

4.9 REFERENCE

4-1 "Study of High-Speed Civil Transports," NASA Contractor Report 4235, Douglas Aircraft Company, New Commercial Programs, Contract NAS1-18378, December 1989.

SECTION 5
ENGINE EMISSIONS

5.1 INTRODUCTION

A major environmental consideration in propulsion technology is engine emissions and the resulting impact on atmospheric ozone. During Phase IIIA, the primary focus was to continue the fleet model fuel burn and annual emissions studies started in the previous system studies, to evaluate updated engine emissions data provided by the engine companies and compare with previously supplied data, and to track and identify any significant trends or factors that could result in lower total emissions per flight.

The reference engine for the previous emissions studies was the Pratt & Whitney (P&W) STF905 variable-stream-control engine (VSCE), for which P&W provided emissions data for several combustor concepts and for variations in engine cycle design parameters. The raw P&W emissions data for this engine were presented in the Phase III report. During the current studies, these data for the STF905 VSCE, along with the corresponding fuel burn data, were extended to a model of total worldwide HSCT fleet annual emissions of those constituents that have been identified as having some possible effect on the environment (e.g., oxides of nitrogen, oxides of sulfur, water vapor, etc.). These data were released to NASA for input into an atmospheric model to evaluate the worldwide atmospheric impact. Fleet model total annual emissions data were developed for three combustor concepts representative of current; near-term; and aggressive, higher risk (far-term) technologies. The data released to NASA and the underlying methodology are summarized in Section 5.2.

Toward the end of Phase IIIA, emissions data were received from both GE and P&W for all engines described in Section 4.2. Some preliminary assessments have been made. Also, areas of further study to reduce total emissions per flight have been identified.

5.2 DEVELOPMENT OF FLEET MODEL TOTAL ANNUAL EMISSIONS

5.2.1 Formation of Oxides of Nitrogen (NO_x)

Oxides of nitrogen (NO_x) have been identified as having the most significant impact on ozone of all the exhaust products of combustion. A key parameter affecting combustion temperature is equivalence ratio (ϕ_i), which is defined as the ratio of the fuel/air ratio to the stoichiometric (ideal combustion) ratio. Since combustion temperatures are maximized at $\phi_i = 1$, so is the production of NO_x, as illustrated in Figure 5-1. Thus, unfortunately, maximum performance tends to maximize NO_x production. Another key parameter in NO_x production is compressor discharge temperature (T_3), which is related to overall pressure ratio (OPR) and is at its design maximum during cruise (as established by compressor material temperature limitations). However, since maximum OPR is required to minimize specific fuel consumption at cruise, this again results in the conflict that maximum performance means maximum NO_x production.

Two combustor concepts being designed by the engine companies to minimize NO_x production by performing the combustion process at ϕ_i values considerably away from unity are also shown in Figure 5-1. In the rich-burn/quick-quench/lean-burn (RB/QQ) concept, all of the fuel is initially consumed in a very rich combustion zone. Because of the lack of oxygen for complete combustion, the temperature, and hence NO_x production rate, in this zone is moderate. The combustion products then pass through a second, lean zone where temperatures are sufficiently high to complete the combustion process, but low enough to prevent accelerated NO_x formation. In between the two zones, large quantities of air are introduced to mix rapidly with the exhaust products of the rich zone to make the mixture lean and to reduce the time at elevated temperature. Initial RB/QQ conceptual studies

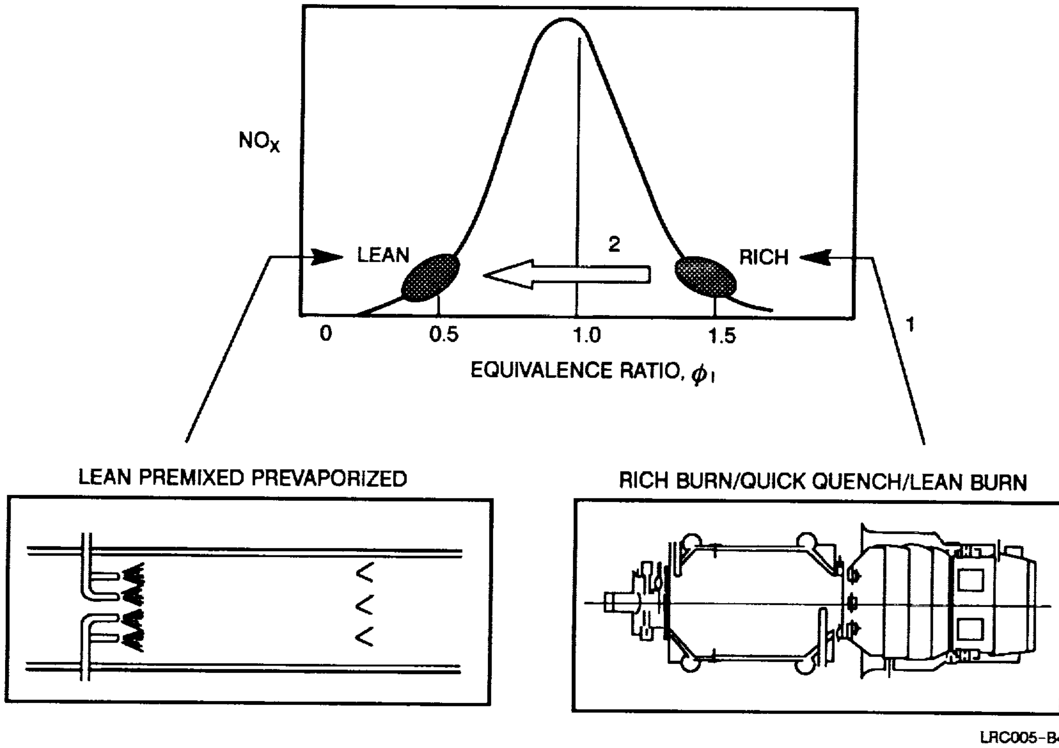


FIGURE 5-1. VARIATION OF NO_X WITH EQUIVALENCE RATIO

have shown promising reductions of total emissions at a very modest increase in aircraft takeoff gross weight (Phase III study).

A second and more technologically aggressive concept is the lean/premixed/prevaporized (PM/PV) combustor, which attempts to achieve low NO_X production rates by not only burning lean, but also by avoiding any localized regions of rich combustion or burning around fuel droplets. This is accomplished by prevaporizing the fuel and injecting it into the air in a premixing zone, which delivers a homogeneous droplet-free mixture to the combustion zone. If these ideal combustor inlet conditions are achieved, this concept is theoretically capable of producing very low NO_X emissions. However, it is considered a high-risk development item when applied to a primary (core) combustor. This is because this concept operates best over a very narrow range of fuel/air ratios, and thus variable-geometry air passages are required to produce the necessary large variations in overall fuel/air ratio associated with large power swings. This would be true for any application, e.g., TBE, VCE, etc. On the other hand, for the duct burner of the P&W VSCE, it is possible to use a two-stage combustor with a premixed pilot stage and a less complex but still advanced-technology high-power stage. The pilot stage would then be sized for cruise at a fixed fuel/air ratio, and variable geometry would not be required. The advanced-technology duct burner high-power stage would be used only during take-off and climb. Thus, development risk would be substantially reduced without a significant increase in total emissions per flight.

Another approach to reducing NO_X production is to reduce cycle OPR, thereby reducing the air temperature entering the combustor. However, this increases engine specific fuel consumption during cruise, which in turn adversely affects aircraft takeoff gross weight (TOGW) and/or range and hence aircraft economics. Some studies of the OPR effect have been performed assuming current-technology combustors. These studies show that the OPR effect provides some benefit toward reducing total emissions per flight. Additional studies are required for advanced-technology combustors.

5.2.2 Development of the Fleet Model

The process used to develop the two-dimensional (2-D) fleet model of total worldwide annual emissions is illustrated in Figure 5-2. The basic procedure is as follows:

- An economic model is developed using the 10 most heavily traveled IATA regions, and a representative mission for each region is generated to produce an altitude-latitude distribution of fuel burn for that region.
- These 10 regions are then weighted by their respective projected travel requirements in relation to the total projected travel for all regions, and then summed to generate a matrix of worldwide fleet total annual fuel burn as a function of latitude and altitude.
- These fleet model fuel burn data are then multiplied by the emissions index for the exhaust product of interest as a function of flight condition to yield a matrix of total worldwide annual emissions as a function of altitude and altitude.

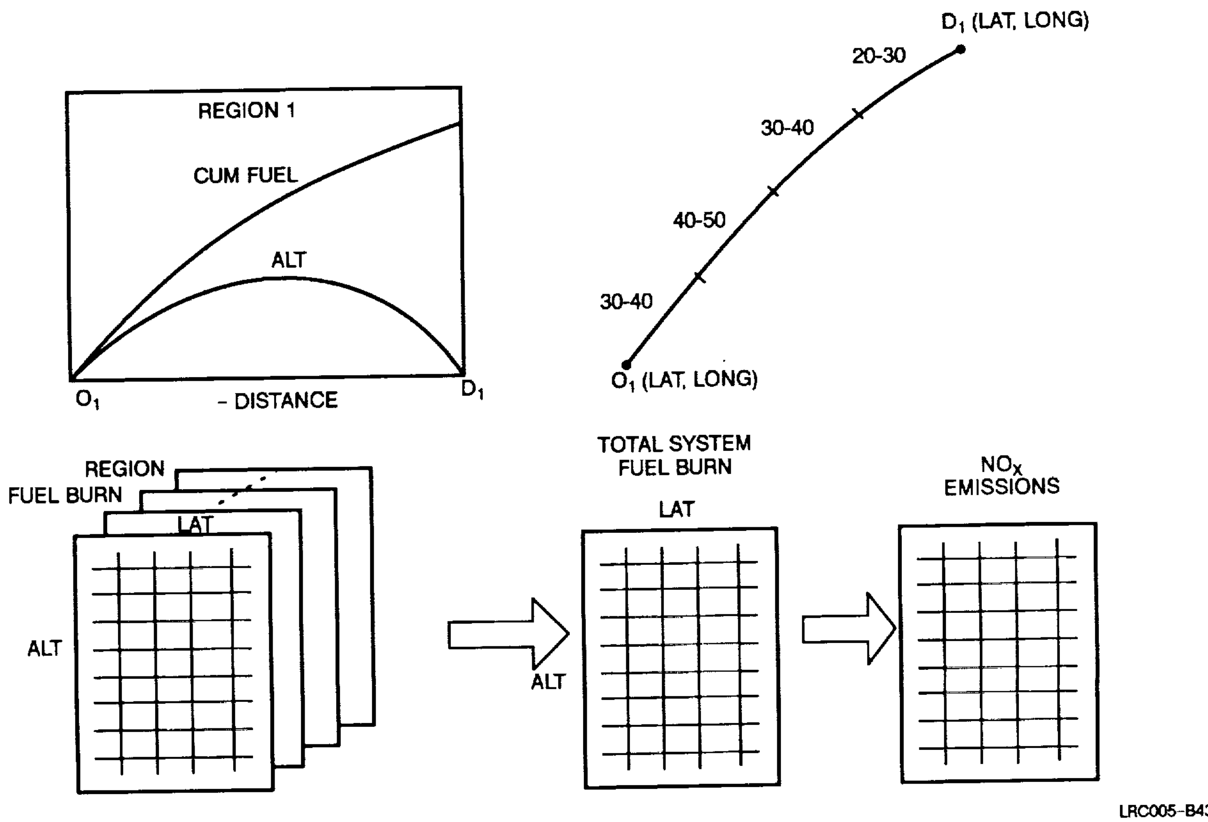


FIGURE 5-2. HSCT FLEET MODELS - NO_X

The resulting total fleet annual fuel burn data for the P&W STF905 VSCE are shown in Table 5-1. Of particular note is that, although it represents a relatively small portion of the total flight segment in terms of time and distance traveled, the fuel burned during climb is more than 30 percent of the total fuel burn. Since climb is generally at maximum-rated conditions with their attendant higher emission levels, this can be of significance to total emissions per flight, as will be discussed later.

5.2.3 Fleet Model Emissions Data Sets

Table 5-2 summarizes the emissions characteristics for the P&W Mach 3.2 STF905 VSCE for several combustor concepts and engine cycle modifications, along with the corresponding weight and specific

REF: NASA-A4

TABLE 5-1
MACH 3.2 FUEL BURN DATA – P&W STF905 VARIABLE-STREAM-CONTROL ENGINE, TSJF FUEL

LATITUDE RANGE (DEG)		TOTAL ANNUAL FUEL BURN (1,000 LB) BY ALTITUDE (KILOMETERS)													
		0.00 TO 2.00	2.00 TO 4.00	4.00 TO 6.00	6.00 TO 8.00	8.00 TO 10.00	10.00 TO 12.00	12.00 TO 14.00	14.00 TO 16.00	16.00 TO 18.00	18.00 TO 20.00	20.00 TO 22.00	22.00 TO 24.00	24.00 TO 26.00	TOTAL ALL ALTITUDES
-90	-80	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-80	-70	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-70	-60	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-60	-50	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-50	-40	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-40	-30	416,119	77,289	77,266	77,266	80,264	101,554	101,510	64,679	57,497	57,497	147,778	437,212	0	1,695,932
-30	-20	698,750	200,732	200,688	200,802	210,722	282,061	281,900	155,336	130,707	130,716	712,619	2,679,857	0	5,884,890
-20	-10	0	0	0	0	0	0	0	0	0	0	1,256,578	2,514,830	0	3,771,408
-10	0	0	0	0	0	0	0	0	0	0	0	1,293,387	2,371,380	0	3,664,767
0	10	1,210,363	297,313	297,313	297,313	314,183	436,916	436,618	232,197	192,505	192,354	541,817	1,013,506	0	5,462,398
10	20	621,492	158,330	158,261	158,316	169,678	231,772	229,645	76,826	3,198	3,199	63,494	1,207,756	0	3,081,967
20	30	90,770	15,431	15,430	15,376	13,475	18,484	20,450	61,269	113,054	113,055	1,144,178	1,353,719	0	2,974,691
30	40	2,010,933	531,455	534,044	534,033	558,479	735,926	735,526	423,821	362,955	362,958	2,710,093	3,110,771	0	12,610,993
40	50	2,404,958	553,196	550,346	551,992	588,731	823,018	822,428	437,034	361,973	361,991	6,044,288	15,000,550	0	28,500,505
50	60	541,726	50,632	50,633	48,998	45,232	45,232	45,231	45,244	45,232	45,231	45,233	8,203,406	0	9,212,033
60	70	0	0	0	0	0	0	0	0	0	0	0	0	0	0
70	80	0	0	0	0	0	0	0	0	0	0	0	0	0	0
80	90	0	0	0	0	0	0	0	0	0	0	0	0	0	0
TOTAL FOR ALL LATITUDES		7,995,112	1,884,378	1,883,981	1,884,096	1,980,765	2,674,964	2,673,307	1,496,407	1,267,121	1,267,001	13,959,466	37,892,986	0	76,859,584

CLIMB FUEL → 25,007,132
CRUISE FUEL → 51,852,452

LRC005-47

TABLE 5-2
SUMMARY OF P&W STF905 VSCE EMISSIONS AND PERFORMANCE CHARACTERISTICS FOR DIFFERENT COMBUSTOR CONCEPTS AND ENGINE CYCLE PARAMETERS

EFFECT OF COMBUSTOR CHANGES								
				PREMIXED, PREVAPORIZED MAIN BURNER WITH:		EFFECT OF CYCLE CHANGES		
PARAMETER	UNITS	BASELINE CONCEPT	RB/QQ MAIN	NEAR-TERM DUCT BURN	PREMIXED DUCT BURN	REDUCED CET	REDUCED OPR	REDUCED OPR AND CET
MAIN COMBUSTOR	TYPE	CURRENT	RICH BURN, QUICK QUENCH	PREMIXED, PREVAPORIZED	PREMIXED, PREVAPORIZED	CURRENT	CURRENT	CURRENT
DUCT BURNER PILOT STAGE	TYPE	CURRENT	CURRENT	CURRENT	PREMIXED	CURRENT	CURRENT	CURRENT
OVERALL PRESSURE RATIO (OPR)	(-)	14.30	14.30	14.30	14.30	14.30	11.45	14.30
COMBUSTOR EXIT TEMP (CET)	°F	3,440	3,440	3,440	3,440	3,140	3,440	3,140
NO_x EMISSIONS INDEX FOR MACH 3.2 CRUISE AT 65,000 FEET	LB NO_x PER 1,000 LB FUEL	39.5	12.1	8.65	6.10	43.0	31.6	33.3
NO_x EMISSIONS INDEX FOR MACH 0.95 CLIMB AT 30,000 FEET	LB NO_x PER 1,000 LB FUEL	8.1	8.1	3.45	3.15	9.5	6.1	6.9
DELTA ENGINE WEIGHT	LB	BASE	+245	+520	+655	-120	+170	+50
DELTA MACH 3.2 CRUISE SFC	PERCENT	BASE	0.0	0.0	0.0	-2.4	+4.1	+2.0
DELTA SUBSONIC CRUISE SFC	PERCENT	BASE	0.0	0.0	0.0	-1.4	+0.9	-0.5

LRC005-B48

fuel consumption (sfc) penalties. Since the VSCE has both a main (core) combustor and a duct burner, several combinations of combustor concepts are possible. The general trend is that as the NO_x emissions index ($EINO_x$) — defined as pounds of exhaust product per 1,000 pounds of fuel burned — is reduced, there is a corresponding increase in combustor weight and hence engine weight, but without any sfc penalties. On the other hand, changing cycle parameters does affect both engine weight and sfc. These cycle perturbations were made at the sea level static (SLS) sizing point, while maintaining a constant ratio of bypass-to-core exhaust velocities of 1.70 to preserve the inverted velocity profile for noise reduction purposes.

Using the emissions data of Table 5-2 and the fuel burn data of Table 5-1, a matrix of emissions data sets was established (Table 5-3). Data sets A3 through A5 for the P&W STF905 VSCE combustor configurations are representative of near-, mid-, and far-term (relatively aggressive) technology levels:

- NASA-A3 — Current-technology main and duct burner combustors
- NASA-A4 — Rich-burn/quick-quench (RB/QQ) main combustor and current-technology duct burner
- NASA-A5 — Premixed/prevaporized (PM/PV) main combustor and premixed duct burner pilot stage

TABLE 5-3
MACH 3.2 EMISSIONS DATA SET SUMMARY

DATA SET	AIRPLANE	CERT YEAR	FLEET	FLEET PROFILE*	FUEL	COMBUSTOR
A3	M3.2	2010	362, 10 REGIONS	ALL SUPERSONIC	JP-7	BASELINE
A4	M3.2	2010	362, 10 REGIONS	ALL SUPERSONIC	JP-7	RBQQ
A5	M3.2	2010	INTL, 10 REGIONS	ALL SUPERSONIC	JP-7	LPP
A6	M3.2/VSCE	2005	INTL, 10 REGIONS	ALL SUPERSONIC	JP-7	RBQQ/LPP
A7	M3.2/TBE	2005	INTL, 10 REGIONS	ALL SUPERSONIC	JP-7	RBQQ
A8	M3.2/VCE	2005	INTL, 10 REGIONS	ALL SUPERSONIC	JP-7	BASELINE
A9	M3.2/FLA	2005	INTL, 10 REGIONS	ALL SUPERSONIC	JP-7	BASELINE

*DATA PROVIDED IN 2-km INCREMENTS FOR ALTITUDE AND 10-DEG INCREMENTS FOR LATITUDE LRC005-B49

Using average $EINO_x$ values for climb and supersonic cruise, worldwide total NO_x emissions for Data Sets NASA-A3, -A4, and -A5 were established. A sample of the NASA-A4 data set is shown in Table 5-4. Table 5-5 shows the total annual NO_x emissions with altitude at all latitudes for the three Mach 3.2 data sets. The reduction in total emissions that can be achieved with advanced combustor concepts is also shown in Table 5-5. The fraction of total fuel burn and corresponding fraction of total NO_x emissions below a given altitude for the NASA-A4 configuration is shown in Figure 5-3.

Estimated changes in aircraft takeoff gross weight (TOGW) for the various concepts are summarized in Figure 5-4. All of data shown are based on takeoff at maximum augmented power conditions, but the trends are the same for other takeoff thrust settings. In general, reductions in $EINO_x$ are accompanied by increases in aircraft TOGW, but the increase in the latter is relatively small, and thus there would be a net reduction in total emissions per flight despite the larger total fuel burns. The reduction in TOGW for reduced combustor exit temperature (CET) reflects the lower engine weight, but at the penalty of increased NO_x emissions due to the increased size of the duct burner to maintain jet velocity ratio at SLS conditions and preserve the inverted velocity profile.

Similar results were obtained by GE for the GE21/F14, Study M1 VCE with current-technology combustors. In the GE study, a reduction in OPR from 19 to 15 resulted in approximately a 25-percent reduction in $EINO_x$ with less than a 3-percent increase in aircraft TOGW. GE needs to conduct similar studies using advanced-technology combustors.

TABLE 5-4
MACH 3.2 NO_X EMISSIONS DATA – P&W STF905 VARIABLE STREAM-CONTROL ENGINE, TSJF FUEL

REF: NASA-A4

– RICH-BURN/QUICK-QUENCH MAIN COMBUSTOR
– CURRENT-TECHNOLOGY DUCT BURNER

NO_X EMISSIONS INDEX (LB PER 1,000 LB FUEL)

CLIMB: 8.10

CRUISE: 12.10

MOLECULAR WT: 32.52 (CRUISE AVERAGE)

LATITUDE RANGE (DEG)	TOTAL ANNUAL EMISSIONS (MOLECULES) BY ALTITUDE (km)													
	0.00 TO 2.00	2.00 TO 4.00	4.00 TO 6.00	6.00 TO 8.00	8.00 TO 10.00	10.00 TO 12.00	12.00 TO 14.00	14.00 TO 16.00	16.00 TO 18.00	18.00 TO 20.00	20.00 TO 22.00	22.00 TO 24.00	24.00 TO 26.00	TOTAL ALL ALTITUDES
-90 -80	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-80 -70	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-70 -60	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-60 -50	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-50 -40	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-40 -30	2,8316E+31	5.2594E+30	5.2578E+30	5.2579E+30	5.4618E+30	6.9107E+30	6.9077E+30	4.4013E+30	3.9126E+30	3.9126E+30	1.5022E+31	4.4444E+31	0	1.3506E+32
-30 -20	4.7549E+31	1.3660E+31	1.3657E+31	1.3664E+31	1.4339E+31	1.9194E+31	1.9183E+31	1.0570E+31	8.8944E+30	8.8951E+30	7.2440E+31	2.7242E+32	0	5.1446E+32
-20 -10	0	0	0	0	0	0	0	0	0	0	1.2774E+32	2.5564E+32	0	3.8338E+32
-10 0	0	0	0	0	0	0	0	0	0	0	1.3148E+32	2.4106E+32	0	3.7254E+32
0 10	8.2364E+31	2.0232E+31	2.0232E+31	2.0232E+31	2.1380E+31	2.9732E+31	2.9711E+31	1.5801E+31	1.3100E+31	1.3089E+31	5.5077E+31	1.0303E+32	0	4.2398E+32
10 20	4.2292E+31	1.0774E+31	1.0769E+31	1.0773E+31	1.1546E+31	1.5772E+31	1.5627E+31	5.2279E+30	2.1764E+29	2.1772E+29	6.4544E+30	1.2277E+32	0	2.5244E+32
20 30	6.1768E+30	1.0500E+30	1.0500E+30	1.0463E+30	9.1696E+29	1.2578E+30	1.3916E+30	4.1693E+30	7.6932E+30	7.6933E+30	1.1631E+32	1.3761E+32	0	2.8636E+32
30 40	1.3684E+32	3.6165E+31	3.6341E+31	3.6340E+31	3.8004E+31	5.0079E+31	5.0052E+31	2.8841E+31	2.4699E+31	2.4699E+31	2.7549E+32	3.1622E+32	0	1.0538E+33
40 50	1.6365E+32	3.7644E+31	3.7450E+31	3.7562E+31	4.0062E+31	5.6005E+31	5.5965E+31	2.9740E+31	2.4632E+31	2.4633E+31	6.1442E+32	1.5249E+33	0	2.6466E+33
50 60	3.6864E+31	3.4455E+30	3.4455E+30	3.3343E+30	3.0780E+30	3.0780E+30	3.0779E+30	3.0788E+30	3.0780E+30	3.0779E+30	4.5981E+30	8.3390E+32	0	9.0406E+32
60 70	0	0	0	0	0	0	0	0	0	0	0	0	0	0
70 80	0	0	0	0	0	0	0	0	0	0	0	0	0	0
80 90	0	0	0	0	0	0	0	0	0	0	0	0	0	0
TOTAL FOR ALL LATITUDES (MOLECULES NO_X)	5.4406E+32	1.2823E+32	1.2820E+32	1.2821E+32	1.3479E+32	1.8203E+32	1.8192E+32	1.0183E+32	8.6226E+31	8.6218E+31	1.4190E+33	3.8519E+33	0	6.9727E+33

CLIMB NO_X → 1.7017E+33 MOLECULES NO_X

CRUISE NO_X → 5.2710E+33 MOLECULES NO_X

LRC005-50

TABLE 5-5
P&W STF905 VARIABLE-STREAM-CONTROL ENGINE COMPARISON OF NO_X EMISSIONS FOR VARIOUS COMBUSTOR CONCEPTS

ALTITUDE RANGE (km)	ANNUAL FUEL BURN (1,000 LB)	TOTAL ANNUAL NO_X EMISSIONS (MOLECULES)		
		BASELINE (NASA-A3)	RB/QQ (NASA-A4)	PM/PV (NASA-A5)
0.00 TO 2.00	7,995,112	5.6266E+32	5.6011E+32	2.1847E+32
2.00 TO 4.00	1,884,378	1.3261E+32	1.3201E+32	5.1492E+31
4.00 TO 6.00	1,883,981	1.3259E+32	1.3199E+32	5.1481E+31
6.00 TO 8.00	1,884,096	1.3259E+32	1.3199E+32	5.1484E+31
8.00 TO 10.00	1,980,765	1.3940E+32	1.3877E+32	5.4126E+31
10.00 TO 12.00	2,674,964	1.8825E+32	1.8740E+32	7.3095E+31
12.00 TO 14.00	2,673,307	1.8813E+32	1.8728E+32	7.3050E+31
14.00 TO 16.00	1,496,407	1.0531E+32	1.0483E+32	4.0890E+31
16.00 TO 18.00	1,267,121	8.9174E+31	8.8770E+31	3.4625E+31
18.00 TO 20.00	1,267,001	8.9165E+31	8.8762E+31	3.4622E+31
20.00 TO 22.00	13,959,466	4.7589E+33	1.4590E+33	7.9513E+32
22.00 TO 24.00	37,892,986	1.2999E+34	3.9707E+33	2.1672E+33
24.00 TO 26.00	0	0	0	0
CLIMB TOTAL	25,007,132	6.3502E+33	2.3366E+33	1.1862E+33
CRUISE TOTAL	51,852,452	1.3167E+34	4.8450E+33	2.4595E+33
TOTAL, ALL ALTITUDES	76,859,584	1.9517E+34 BASE	7.1816E+33 36.8% OF BASE	3.6457E+33 18.7% OF BASE

BASELINE COMBUSTOR (NASA-A3) ⟶ PRIMARY COMBUSTOR: CURRENT TECHNOLOGY; DUCT BURNER: CURRENT TECHNOLOGY ⟶ CRUISE $EINO_X$ = 39.5

RB/QQ COMBUSTOR (NASA-A4) ⟶ PRIMARY COMBUSTOR: RICH BURN/QUICK QUENCH; DUCT BURNER: CURRENT TECHNOLOGY ⟶ CRUISE $EINO_X$ = 12.1

PM/PV COMBUSTOR (NASA-A5) ⟶ PRIMARY COMBUSTOR: PREMIXED, PREVAPORIZED; DUCT BURNER: PREMIXED ⟶ CRUISE $EINO_X$ = 6.10

NOTE: ALL DATA ARE TOTAL FOR ALL LATITUDES IN ALTITUDE RANGE

LRC005-B51

5.3 RESULTS OF ENGINE AND FLEET EMISSIONS STUDIES

5.3.1 Engine Emissions

Engine performance and emissions data were received from both Pratt & Whitney and General Electric for all Mach 3.2 engine concepts described previously. Although both engine companies provided emissions estimates, there was one distinct difference between the data received from the two engine companies:

- All P&W data assumed use of a rich-burn/quick-quench main combustor and, for the VSCE, an advanced high-power stage and a premixed pilot stage for the duct burner.
- All GE data assumed current-technology combustors. The low-NO_x combustor technology emissions data will be produced in the next assessment phase.

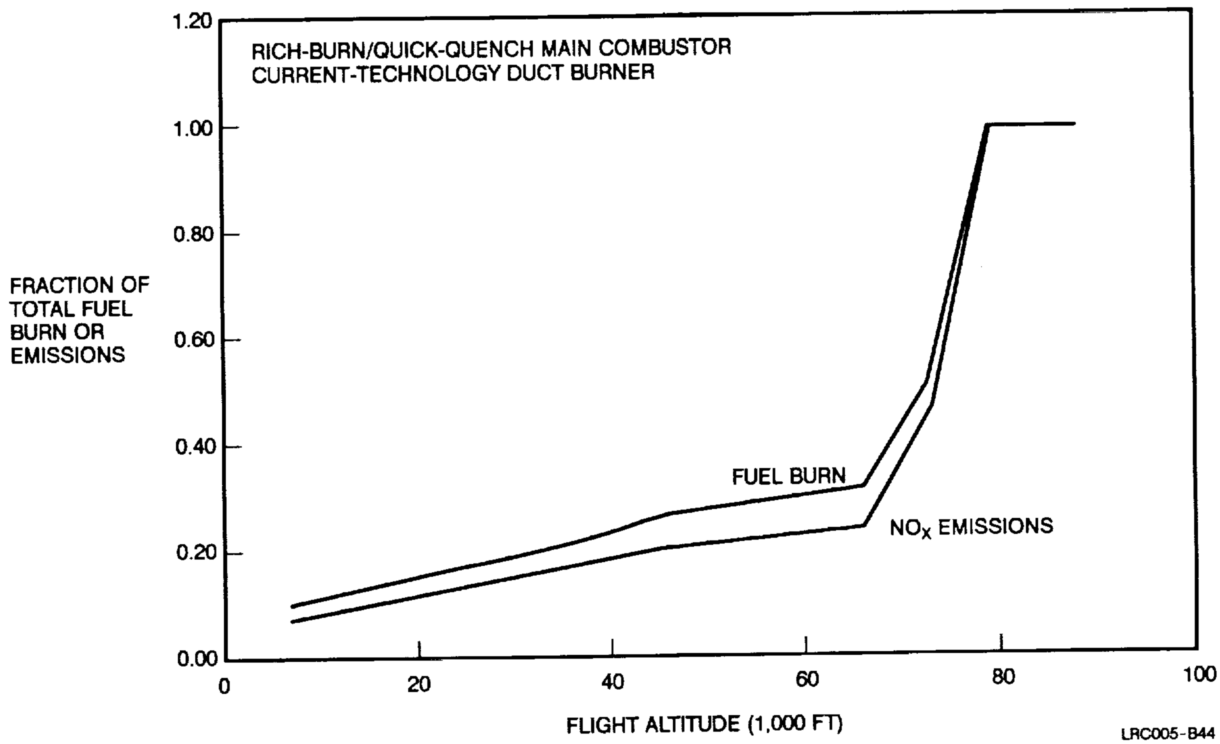


FIGURE 5-3. P&W STF905 VARIABLE-STREAM-CONTROL ENGINE DISTRIBUTION OF ANNUAL NO_X EMISSIONS

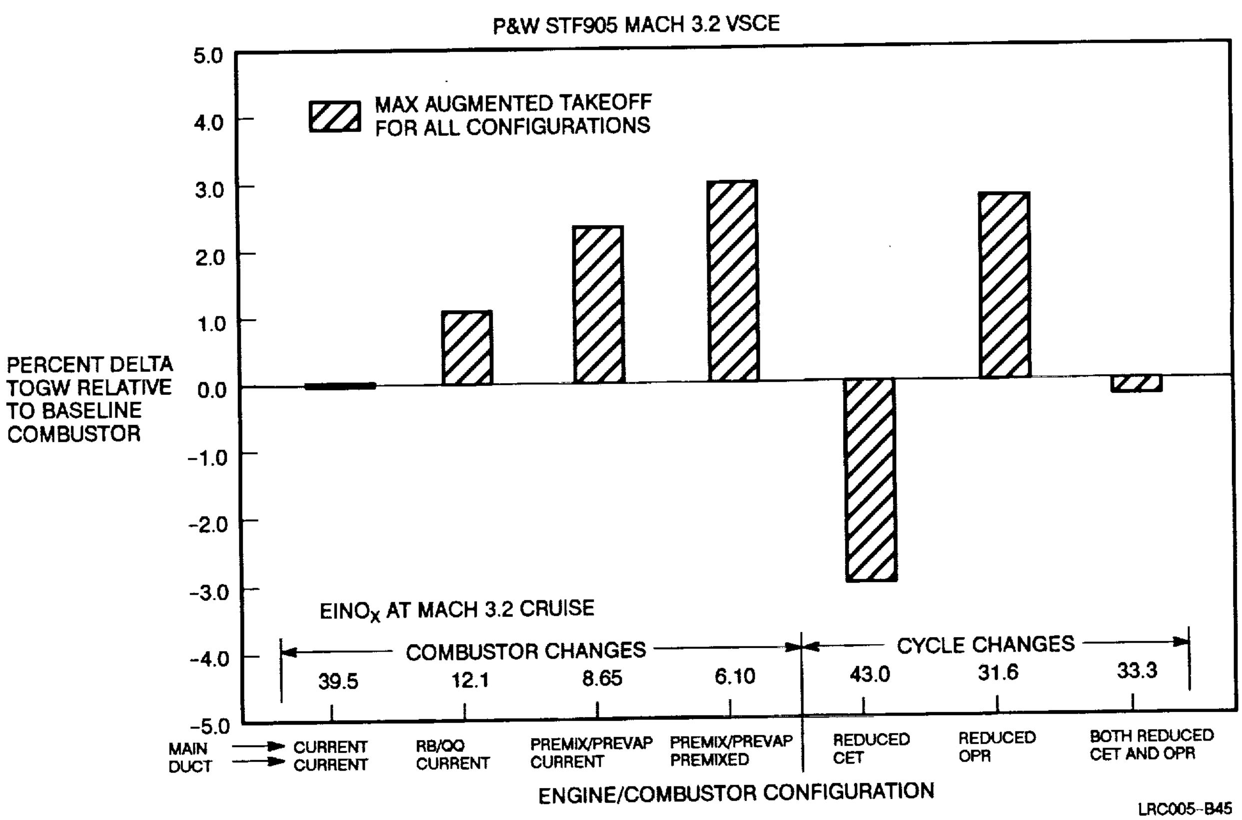


FIGURE 5-4. EFFECT OF LOW-NO_X ENGINE ON TOGW

The NO_x emissions data for the P&W and GE engines at subsonic and supersonic cruise conditions are summarized in Table 5-6. Table 5-7 tabulates NO_x data for the STF947 VSCE for a range of take-off, climb, and cruise flight conditions and power settings.

TABLE 5-6
SUMMARY NO_X EMISSION INDEX DATA FOR ALL PHASE IIIA MACH 3.2 ENGINES

A. PRATT & WHITNEY WITH RICH-BURN/QUICK-QUENCH MAIN COMBUSTOR:
(STF947 VSCE HAS PREMIXED DUCT BURNER PILOT STAGE)

ENGINE DESIGNATION (DESIGN MACH NO.)	ALTITUDE (FT)	MACH NUMBER	PERCENT MAX RATED THRUST	$EINO_X$
STF947 VSCE	30,000	0.95	37.0	1.5
(MACH 3.2)	30,000	0.95	100.0	2.1
	70,000	3.20	36.0	5.6
	70,000	3.20	100.0	16.2
STJ950 TBE	30,000	0.95	46.0	1.8
(MACH 3.2)	30,000	0.95	100.0	3.6
	70,000	3.20	64.0	5.1
	70,000	3.20	100.0	7.8

B. GENERAL ELECTRIC WITH CURRENT-TECHNOLOGY COMBUSTOR

ENGINE DESIGNATION	ALTITUDE (FT)	MACH NUMBER	PERCENT MAX RATED THRUST	$EINO_X$
GE21/F14, STUDY M1,	30,000	0.95	54.0	4.0
VARIABLE-CYCLE ENGINE	30,000	0.95	100.0	5.3
	70,000	3.20	70.0	29.9
	70,000	3.20	100.0	35.1
GE21/FLA1, STUDY A1,	30,000	0.95	56.0	4.0
HIGH-FLOW FAN VCE	30,000	0.95	100.0	6.6
	70,000	3.20	73.0	29.8
	70,000	3.20	100.0	33.8

LRC005-52

The variation of $EINO_x$ with climb Mach number for the P&W STF947 VSCE at maximum augmented thrust, consistent with the aircraft sizing studies, is shown in Figure 5-5. Shown also is the average climb value used for the NASA-A4 studies with a comparable main combustor configuration but with a current-technology duct burner. Figure 5-6 presents $EINO_x$ data for the STF947 VSCE as a function of the percentage of maximum-rated thrust compared with the average value used to develop the NASA-A4 data set. Noted also are significant engine operating points at the corresponding fraction of maximum-rated thrust. As can be seen, this engine is designed to cruise in the

TABLE 5-7
P&W MACH 3.2 STF947 VARIABLE-STREAM-CONTROL ENGINE – NO_X EMISSIONS INDEX VERSUS THRUST AND MACH NUMBER

RICH-BURN/QUICK-QUENCH MAIN COMBUSTOR
PREMIXED DUCT BURNER PILOT STAGE

FLIGHT CONDITION	ALTITUDE (FT)	MACH NUMBER	PERCENT MAX RATED THRUST	$EINO_X$	UNSCALED ENGINE NET THRUST (LB)	
					UNINSTALLED	INSTALLED
MAX RATED CLIMB	0	0.00	100.0	3.1	65,285	65,285
	20,000	0.60	100.0	2.3	35,451	34,940
	30,000	0.95	100.0	2.1	31,897	30,606
	50,000	2.00	100.0	4.8	34,349	32,879
	70,000	3.20	100.0	16.2	28,614	28,734
SUPERSONIC CRUISE:						
MAX DRY:	70,000	3.20	27.0	6.8	7,726	6,598
TSFC BUCKET (UNINS):	70,000	3.20	36.0	5.6	10,301	9,517
PARTIAL AUGMENTED:	70,000	3.20	52.0	4.4	14,879	14,486
MAXIMUM AUGMENTED:	70,000	3.20	100.0	16.2	28,614	28,734
SUBSONIC CRUISE:						
PARTIAL DRY:	30,000	0.95	28.0	1.2	8,931	7,624
TSFC BUCKET (UNINS):	30,000	0.95	37.0	1.5	11,802	10,368
MAX DRY:	30,000	0.95	47.0	1.9	14,992	12,057
MAXIMUM AUGMENTED:	30,000	0.95	100.0	2.1	31,897	30,606

LRC005-B53

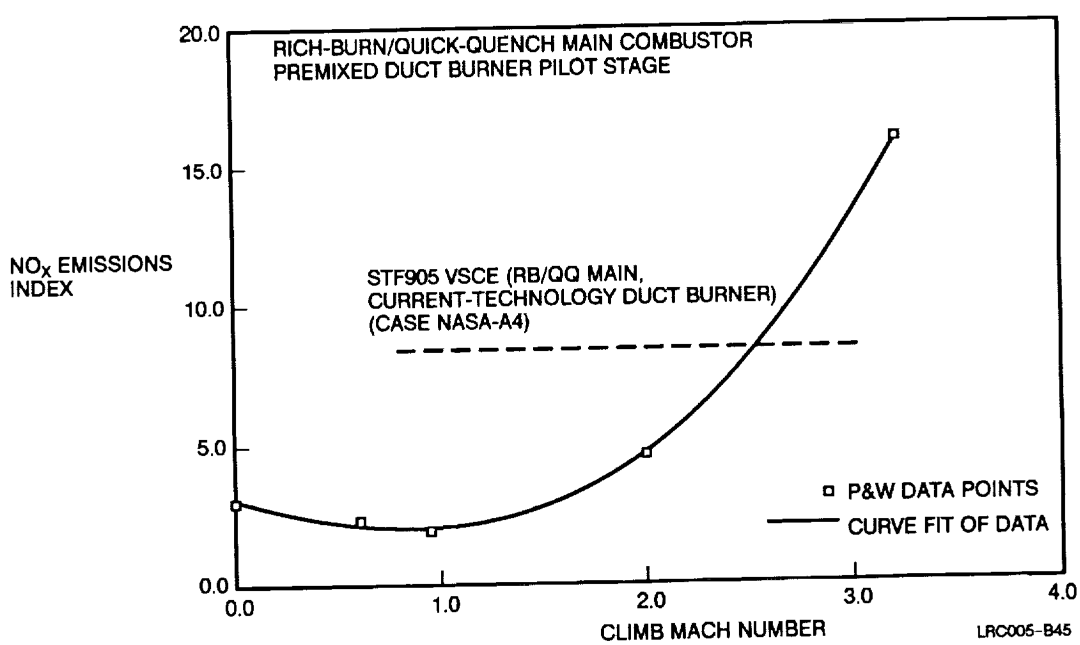


FIGURE 5-5. P&W STF947 VSCE $EINO_X$ AT MAXIMUM CLIMB

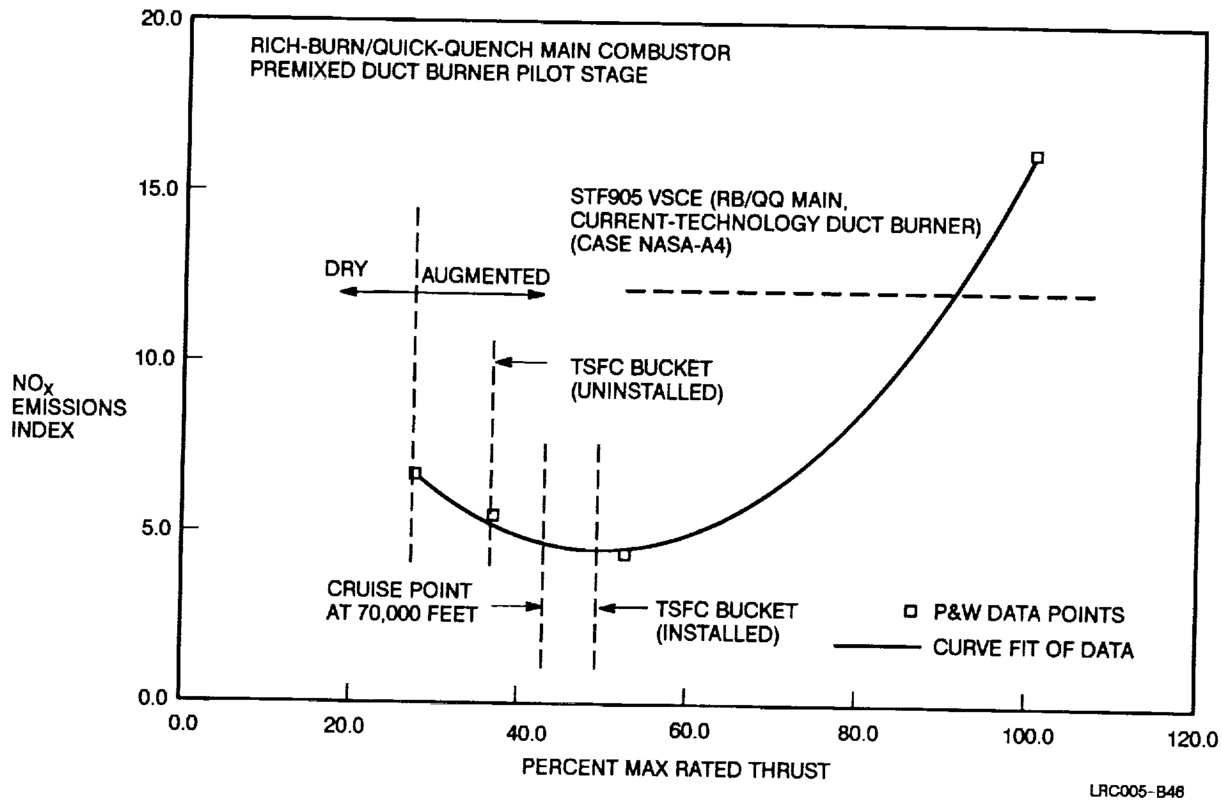


FIGURE 5-6. P&W STF947 VSCE $EINO_x$ AT MACH 3.2 CRUISE

augmented regime, and hence the duct burner design affects engine emissions during cruise. Therefore, although both the STF905 (NASA-A4) and STF947 engines utilize a rich-burn/quick-quench main combustor, the duct burner for the STF947 is equipped with a premixed pilot stage. This accounts for the greatly reduced emissions at power settings around the sfc "bucket." As the duct burner pilot stage emissions are very low, total engine $EINO_x$ decreases with augmentor operation as the ratio of duct burner to main combustor fuel flow increases, then increases as the duct burner high-power stage phases in. Thus, it is critical to match this particular engine cycle with the airframe requirements to ensure that both sfc and $EINO_x$ are minimized during cruise.

Combustor characteristics for single-combustor engines (e.g., turbine bypass) are expected to be similar to those of the two-stage combustor concepts, where the low-power/low-NO_x stage would satisfy cruise requirements and the high-power stage would be used only for climb. Although there may not be an $EINO_x$ bucket there could be an "$EINO_x \times$ sfc" bucket that minimizes total emissions per flight, which is really the design objective relative to engine emissions. This area requires further study, particularly since cruise emissions can be as much as 75 percent of the total emissions for the flight.

5.3.2 Fleet Model Total Annual Emissions

Annual total worldwide fuel burn models for the study engines have continued being developed consistent with the NASA-A4 data set. Additionally, the relative total emissions from analysis of fuel burn data for a representative design mission have been estimated. Table 5-8 compares the percentages of total fuel burn and NO_x emissions during climb for the fleet model with those for the single design mission, using the time history from the STF905 design mission (6,500 nautical miles, supersonic overland cruise). The percentages are not the same, with the fleet model predicting higher percentages during climb, primarily because the fleet model includes ranges shorter than design goals

TABLE 5-8
COMPARISON OF FUEL BURN AND EMISSIONS DISTRIBUTION FOR FLEET MODEL VERSUS INDIVIDUAL DESIGN MISSION

CASE DESCRIPTION	PERCENT OF TOTAL FUEL BURN OR NO_X DURING CLIMB	
	FUEL BURN	TOTAL NO_X
STF905 VSCE FLEET MODEL	32.4	24.4
STF905 VSCE DESIGN MISSION (767,000-LB TOGW; 6,500-N-MI RANGE)	25.5	19.0
STF947 VSCE DESIGN MISSION (900,000-LB TOGW; 5,790-N-MI RANGE)	27.9	23.0

NOTES:

1. STF905 VSCE IS EQUIPPED WITH RICH-BURN/QUICK-QUENCH MAIN COMBUSTOR AND CURRENT-TECHNOLOGY DUCT BURNER
2. STF947 VSCE IS EQUIPPED WITH RICH-BURN/QUICK-QUENCH MAIN COMBUSTOR AND PREMIXED DUCT BURNER PILOT STAGE

LRC005-B53

and subsonic flight data. However, there is sufficient agreement to use this approach for screening purposes. Table 5-8 also shows the corresponding single-mission data for the STF947 VSCE, which are in very good agreement with those for the STF905 design mission. This despite performance differences between the two VSCEs, which resulted in heavier aircraft takeoff gross weight and reduced range for aircraft powered by the STF947.

The cumulative NO_X emissions with aircraft range for the P&W STF905 and STF947 VSCEs and their respective design missions are compared in Figure 5-7. The most significant item is the impact of a premixed pilot stage on the STF947 duct burner, which has reduced total emissions by almost 50 percent despite the aircraft's burning more fuel. In addition, it is recommended that the potential

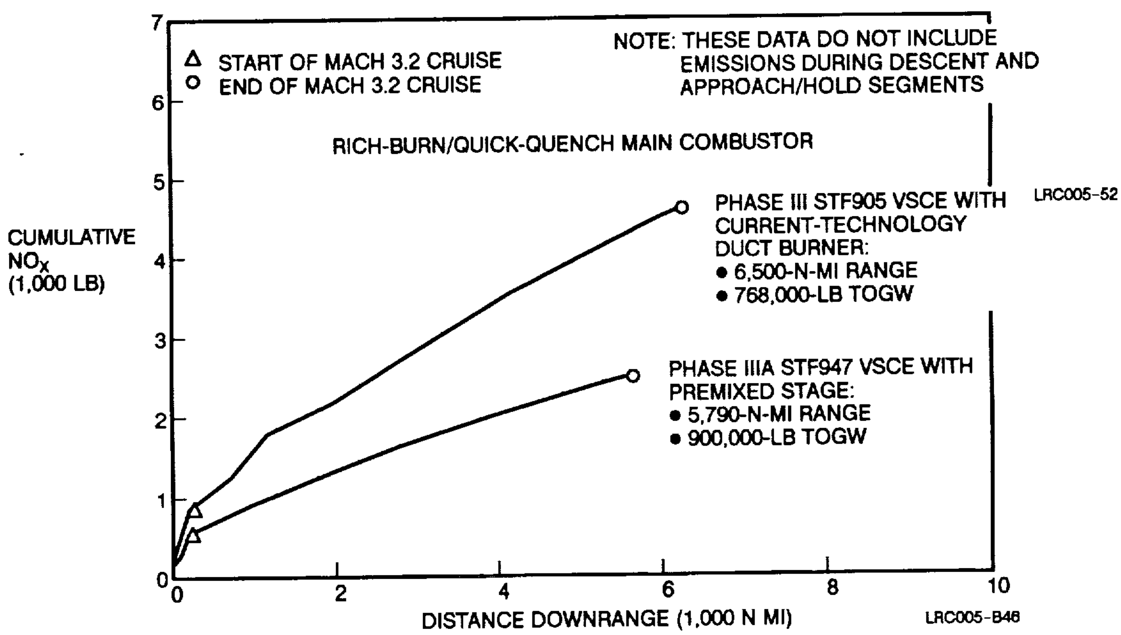


FIGURE 5-7. P&W VSCE NO_X EMISSIONS PER FLIGHT

benefits of lowering the combustor inlet temperature by 100°F be investigated. It is important to look at climb $EINO_x$ values at several points along the climb profile rather than simply using the average value selected.

5.4 CONCLUSIONS AND RECOMMENDATIONS

On the basis of a preliminary assessment of the engine emissions data, several trends have been identified relative to reducing total HSCT emissions:

- The RB/QQ combustor concept offers promise for considerable reduction of total emissions at a very modest increase in aircraft takeoff gross weight. Also, it can be made available in the time frame consistent with HSCT development schedules and aircraft certification goals according to P&W. The RB/QQ also offers slightly less technical risk, as judged by P&W, than the premix/prevaporized concept.
- Assuming a RB/QQ primary (core) combustor in both cases, the addition of a premixed pilot stage to an advanced-technology burner reduces the total emissions for the P&W STF947 VSCE to approximately one-half of those for the STF905 VSCE.
- Similar GE low-NO_x combustor studies should be evaluated.

SECTION 6
LAMINAR FLOW CONTROL (LFC)

6.1 INTRODUCTION

A preliminary engineering integration design study of LFC was conducted, leading to an economic assessment. LFC, with its associated reduction in drag and additional weight and complexity, was applied to a fully turbulent version of the D3.2-3A configuration (see appendix). To quantify the achievable benefit, the method of laminarization chosen was boundary layer suction through a perforated wing skin. Three configuration concepts were addressed. Two of the configurations used the -3A planform and thickness distribution with suction (1) outside the fuel tank boundary and forward of the control surface hinge line (partial LFC), and (2) forward of the control surface hinge line (full LFC). The two configurations are illustrated in Figure 6-1. The third configuration studied was an all supersonic leading edge wing planform with full LFC. A comprehensive study was conducted involving an aerodynamic redesign of the -3A wing, suction system design, material selection, structural design, integration of the suction system, weight assessment, and mission performance evaluation. The final assessment of the LFC configurations was an economic evaluation based on the results of the engineering analysis.

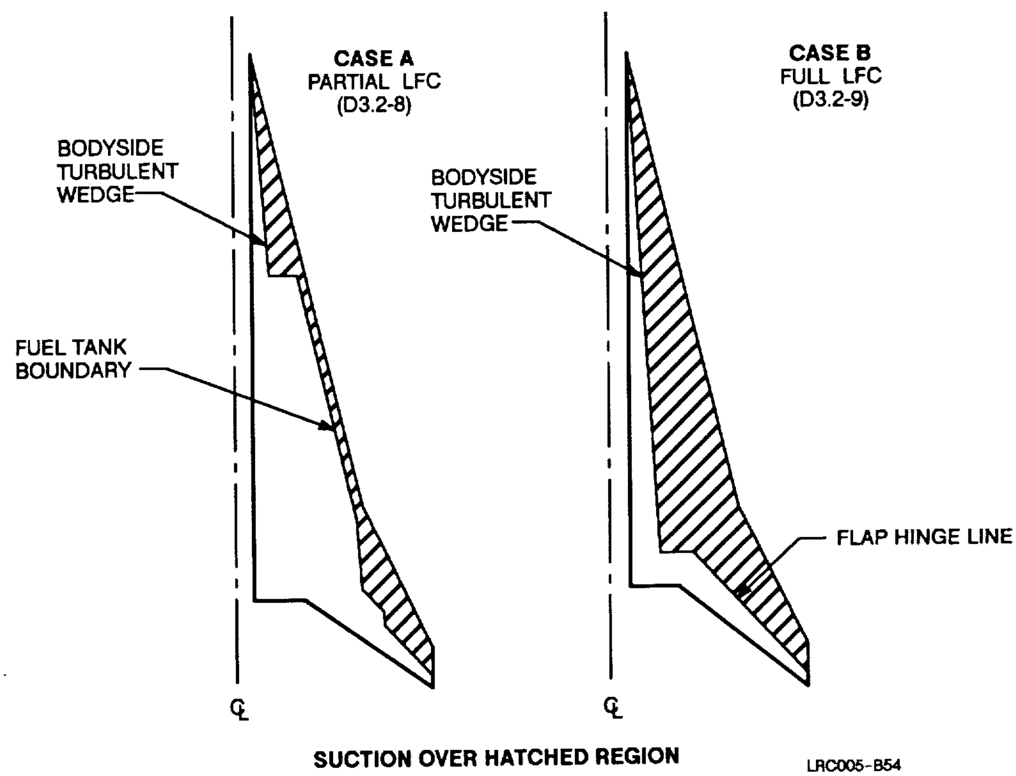


FIGURE 6-1. HSCT D3.2-8 AND - 9 PLANFORM

6.2 AERODYNAMIC DESIGN AND CHARACTERISTICS

A new wing section design was required based on an analysis of the baseline wing (D3.2-3A) at the cruise point, using the Euler code FLO67 (Reference 6-1), which indicated that the wing surface pressures were inappropriate for laminarization. Since the suction system weight is directly related to the amount of suction required, the objective of the LFC design was to minimize the amount of suction required to laminarize the wing. This was achieved by generating wing sections that deliver a pressure distribution with minimum chordwise pressure gradient. For the redesign, the wing was

divided into two regions, the inboard subsonic leading edge (LE) panel, and the supersonic LE panel outboard of the LE break. The baseline wing planform, thickness, and spanwise lift distribution were maintained in the wing redesign procedure.

The inboard wing panel was modified by increasing the leading edge radius to reduce the chordwise extent of the positive pressure gradient in this region and to reduce the susceptibility to transition caused by the instability of stationary cross-flow waves in the boundary layer (BL). This instability is caused by inflections in the BL velocity profile, typically requiring large amounts of suction to suppress. It is for this reason that the baseline pressure distribution was inappropriate for laminarization, and that the redesigned wing represents a significant improvement. A comparison of wing chordwise pressure distributions is presented in Figure 6-2. An overshoot in the pressure distribution near the leading edge followed by a constant velocity region aft is the key to minimizing the amount of suction required to laminarize the new wing.

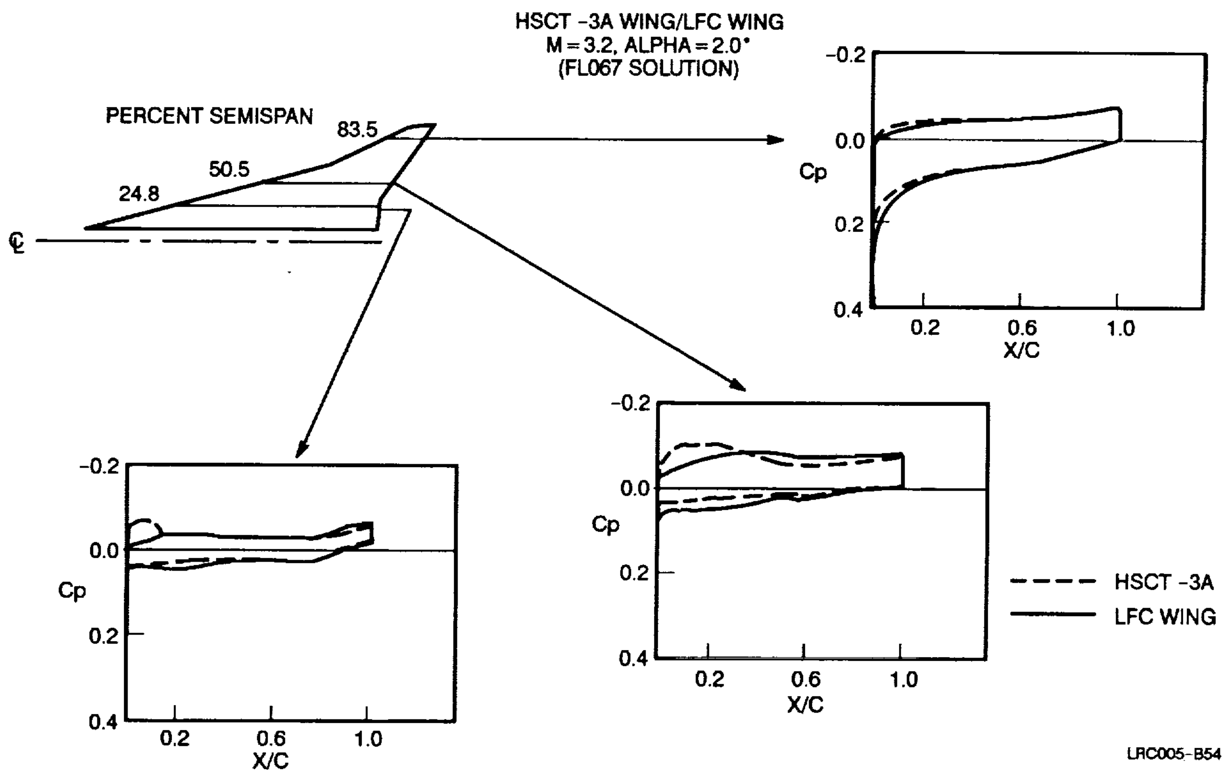


FIGURE 6-2. COMPARISON OF CHORDWISE PRESSURE DISTRUBUTIONS

For the supersonic leading edge panel, the design philosophy was to reduce the leading edge radius and eliminate surface curvature to minimize the chordwise pressure gradient. As the data for 83.5-percent semispan in Figure 6-2 indicate, this objective was not achieved. The three-dimensional flow field effect generated by the inboard panel virtually nullified the purpose of the airfoil redesign in the outboard region.

A comparison of the baseline and new airfoils for the LFC configurations is presented in Figure 6-3.

The two suction pattern concepts persued were illustrated previously in Figure 6-1. The Case A suction pattern or partial LFC concept provides suction outside the fuel tank boundary and forward of the control surface hinge line. Case B provides full-chord LFC, with suction from the leading edge aft to the control surface hinge line. The configurations for Cases A and B were designated D3.2-8

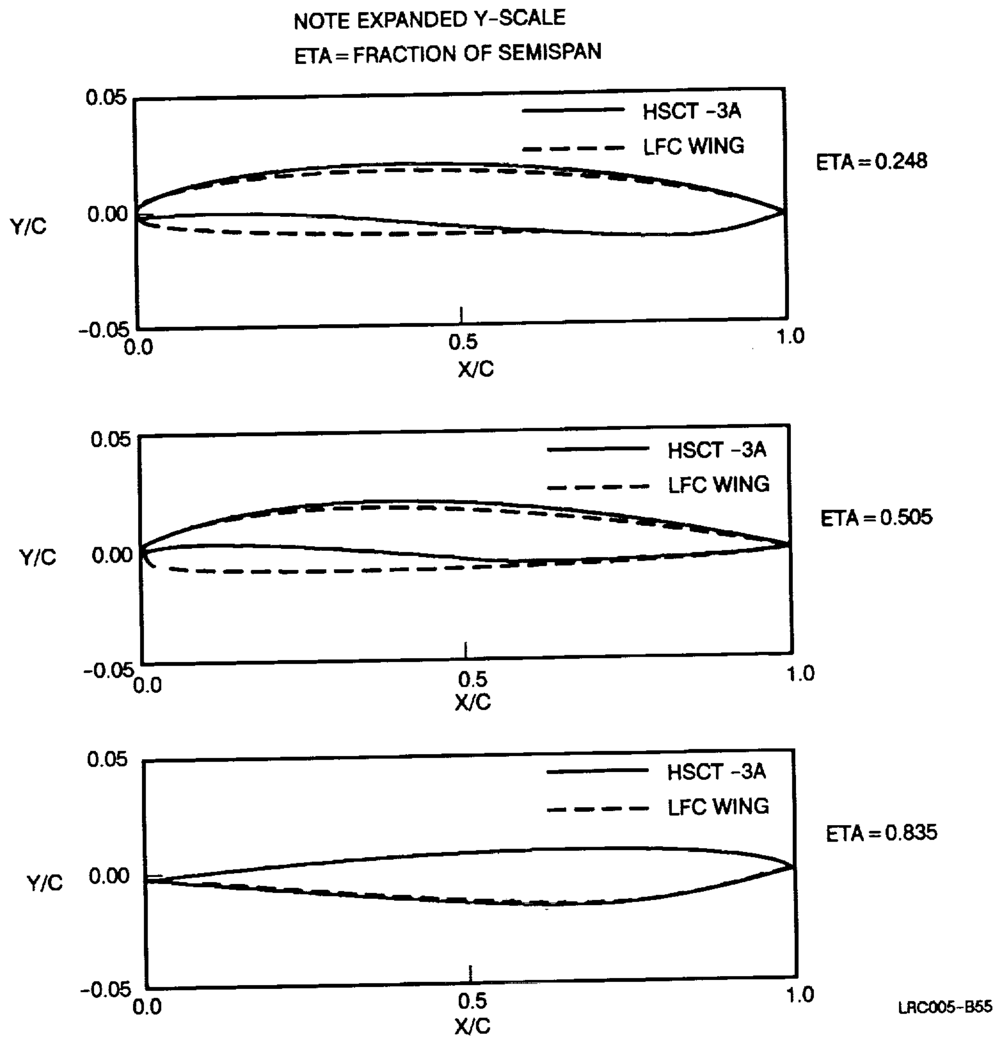


FIGURE 6-3. AIRFOIL COMPARISON OF HSCT D3.2-3A AND LFC WING

and D3.2-9, respectively. Both concepts have no suction over a 4.5-degree wedge (Reference 6-2) at the side of body, starting at the exposed wing apex. This no-suction area corresponds to turbulence that originates on the fuselage, which is not laminarized.

Boundary layer stability analyses were performed using the NASA MARIA (Reference 6-3) and COSAL (Reference 6-4) codes with inputs from the Cebeci and Kaups conical boundary layer program (Reference 6-5). Following the stability analyses, minimum suction levels were determined that would ensure laminar flow over Patterns A and B. The suction levels are presented in Figure 6-4 in terms of the suction coefficient Cq, the ratio of mass flux through the skin to the free-stream mass flux (negative values indicate flow into the wing). The suction levels for Cases A and B are the same, only the chordwise extent differs. Based on the suction levels required, air captured at the adiabatic recovery temperature of the free-stream flow must be removed from the wing boundary layer at a rate of 17.3 lbm/s for Case A (4,100 ft^2 of suction area) and 25.2 lbm/s for Case B (9,600 ft^2 of suction area) at the initial cruise altitude of 65,700 feet. In both cases, the suction flow is Mach 0.4 in the duct.

Using the results from a far-field wave drag analysis generated by HABP (Reference 6-6), the fuselage was area-ruled for the new wing geometry. The baseline configuration and the new LFC designs were then analyzed using a marching solution Euler. A comparison of the inviscid results for the fully

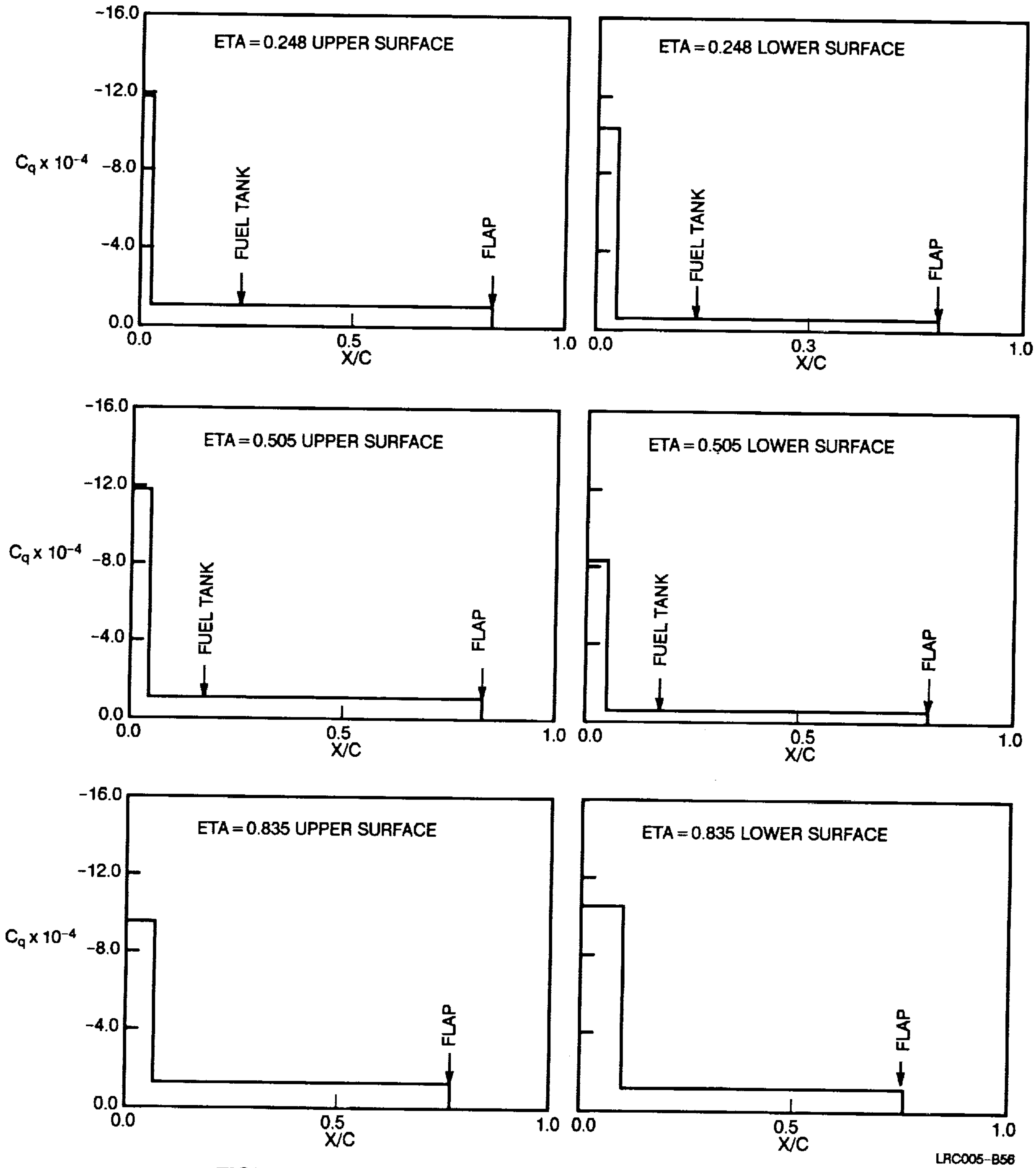


FIGURE 6-4. CHORDWISE SUCTION DISTRUBUTIONS

turbulent, partial LFC and full LFC configurations indicated that there was no drag-due-to-lift penalty for the new leading edge geometry at the cruise point. The Euler code was necessary as no in-house linear theory code could resolve the geometrical differences between the baseline and the new wing.

An assessment of the skin friction drag reduction for the two LFC cases indicated a 7-percent reduction in total drag for Case A and an 11-percent reduction for Case B. Drag polars for the three configurations (fully turbulent, partial LFC, full LFC) are presented in Figure 6-5. D3.2-7 is the designation for the fully turbulent version of the D3.2-3A configuration, i.e., same airfoil sections as the D3.2-3A.

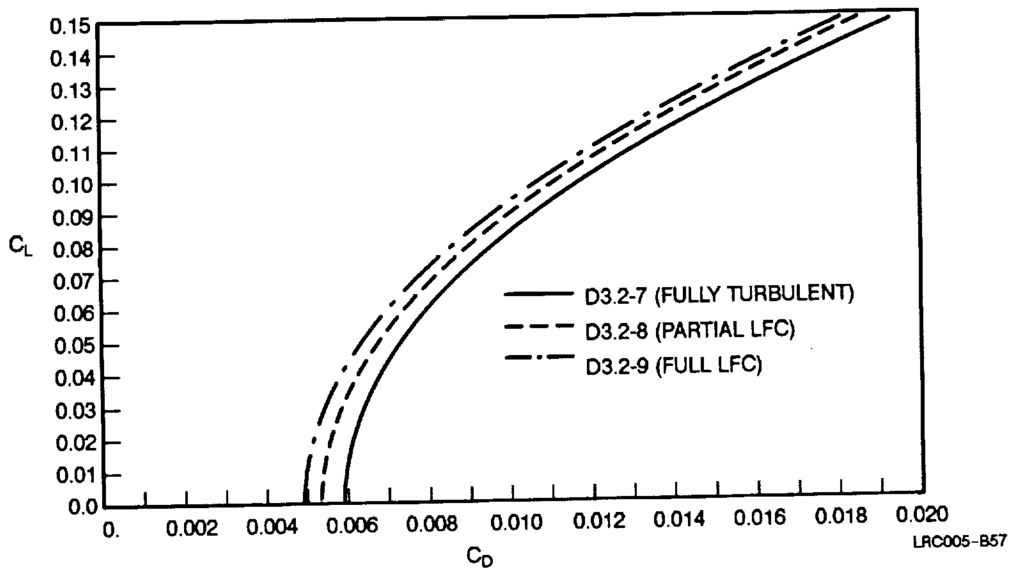


FIGURE 6-5. EFFECT OF LFC AND CORRESPONDING WING DESIGN ON MACH 3.2 DRAG POLARS

6.3 SUCTION SYSTEM POWER REQUIREMENT AND WEIGHT

Compressor sizing data for the full LFC system are summarized in Table 6-1. The compressors have been sized with separate systems for the upper and lower surfaces due to different flow rates and pressures. Although compressor-corrected airflows and pressure ratios are essentially independent of cruise altitude, the power requirements are not, and hence the system has been sized for start of

TABLE 6-1
LFC COMPRESSOR SIZING DATA (FULL LFC SYSTEM)

ITEM	UNITS	UPPER SURFACE	LOWER SURFACE
TOTAL FLOW PER SIDE	LB/SEC	6.40	6.20
INLET TOTAL PRESSURE	LB/FT^2	66.56	122.58
INLET TOTAL TEMPERATURE	°F	470	490
CORRECTED AIRFLOW	LB/SEC	273	145
PRESSURE RATIO	----	3.421	1.857
COMPRESSOR TYPE	----	CENTRIFUGAL	CENTRIFUGAL
NO. OF COMPRESSORS PER SIDE	----	2	2
MAXIMUM DIA/AXIAL LENGTH	IN.	93/24	64/15
INLET/OUTLET CONNECTION	IN.	36/16	26/15
COMPRESSOR WEIGHT (EACH)	LB	1,000	350
ELECTRIC DRIVE SYSTEM (MOTORS, SHAFTS, ETC.) WT	LB	523	234
MOUNTING (SUPPORTS, ATTACHMENTS, ETC.) WT	LB	125	44
SHAFT POWER (EACH)	HP	560	250

SIZING POINT: START OF CRUISE (MACH 3.2 AT 65,700 FEET)
INLET TOTAL TEMPERATURE = SKIN TEMPERATURE

LRC005-B68

cruise, or Mach 3.2 at 65,700 feet. The compressor sizes, weights, and power requirements were obtained from a recognized potential supplier based upon the system requirements data shown. System requirements were based on the computed suction requirements, average wing skin pressures, and a system exit Mach number of 1.0. This exit condition minimizes power required (Reference 6-7) and hence compressor weight. Because of specific speed considerations, centrifugal compressors have been selected, with the relatively large size dictating two compressors per side (i.e., left and right), per surface, resulting in a total of eight compressors for the airplane.

For the sizing studies, it has been assumed that the compressors are driven by electric motors (electric power from generators used to drive aircraft systems) and are located in the lower aft fuselage area. The penalty associated with the electric power is reflected in the drive system weight. An efficiency of 0.95 was assumed for the electric drive motors. Using the data in Table 6-1, the total LFC suction system weight and power requirements are summarized in Table 6-2. The data in Table 6-2 do not include the weights for the suction surface (outer wing skin or leading edge surface), suction plenums, flutes, or seals, as these are considered part of the aircraft structure. The system totals are:

- Partial LFC: 6,370 pounds and 2,240 horsepower (560 horsepower/engine)
- Full LFC: 9,100 pounds and 3,240 horsepower (810 horsepower/engine)

TABLE 6-2
LFC SUCTION SYSTEM WEIGHTS

ITEM	UNITS	PARTIAL LFC SYSTEM	FULL LFC SYSTEM
TOTAL FLOW PER SIDE (BOTH SURFACES)	LB/SEC	8.65	12.60
COMPRESSORS, UPPER SURFACE	LB	2,800	4,000
COMPRESSORS, LOWER SURFACE	LB	980	1,400
ELECTRIC DRIVE SYSTEM	LB	2,120	3,030
MOUNTINGS, ETC.	LB	470	670
TOTAL WEIGHT (LESS STRUCTURE, PIPING/DUCTING, LE PLENUMS, FLUTES, ETC.)	LB	6,370	9,100
SHAFT POWER (PER ENGINE)	HP	560	810

LRC005-B69

Note that the LFC system is only turned on for the cruise portion of the aircraft's mission, and the above required power data are for start of cruise. At end of cruise (Mach 3.2 at 76,900 feet) the power requirements are approximately 40 percent less, with the average cruise values then being approximately 1,790 horsepower and 2,590 horsepower (approximately 450 and 650 horsepower/engine) for the partial and full LFC systems, respectively.

6.4 STRUCTURAL DESIGN, MATERIALS, AND LFC DUCTING

A materials study was conducted to select an LFC skin and substructure design and to define the structural-material concepts that will result in a minimum weight structure that meets the aerodynamic requirements. The structural-material concepts that were analyzed were skin-stringer; superplastic-formed/diffusion-bonded; and honeycomb structures utilizing advance metal matrix composites. In evaluating these concepts, consideration was given to fail-safe design, damage tolerance, maintainability, and producibility.

The Federal Aviation Regulations (FAR) administered by the Federal Aviation Administration (FAA) place primary aircraft structure into two categories: damage-tolerant structures and safe-life structures. The damage-tolerant structure must be able to fly safely even after it has experienced partial or complete failure. This category includes almost every piece of primary structure on the airframe. The FAR requires that the evaluation of these structures must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage (Reference 6-8). The design of current LFC structures uses the damage-tolerant design concept because it is more applicable in all practical respects compared to the safe-life design concept.

The design criteria used in the analysis are presented in Table 6-3. Load factors of +2.5 and 1.0, and factors of safety of 1.5 are specified. A typical flight profile for a Mach 3.2 aircraft is shown in Figure 6-6. The design loads were developed from the three flight conditions shown in Table 6-4.

TABLE 6-3
DESIGN CRITERIA FOR MACH 3.2

LOAD FACTOR • SUBSONIC: +2.5, -1.0 • SUPERSONIC: +2.5, -1.0	DESIGN LIFE 60,000 FLIGHT HOURS
MAXIMUM Q = 1,100 PSF (SUPERSONIC) = 600 PSF (SUBSONIC)	DESIGN GAG LANDINGS 20,000 FLIGHTS FACTOR OF SAFETY = 1.5
LANDING SINK SPEED • 10 FPS (EMPTY) • 6 FPS (FULL FUEL)	LANDING GEAR DEPLOYMENT SPEED 260 KNOTS
	THERMAL ENVIRONMENT -65°F TO 680°F

LRC005-B70

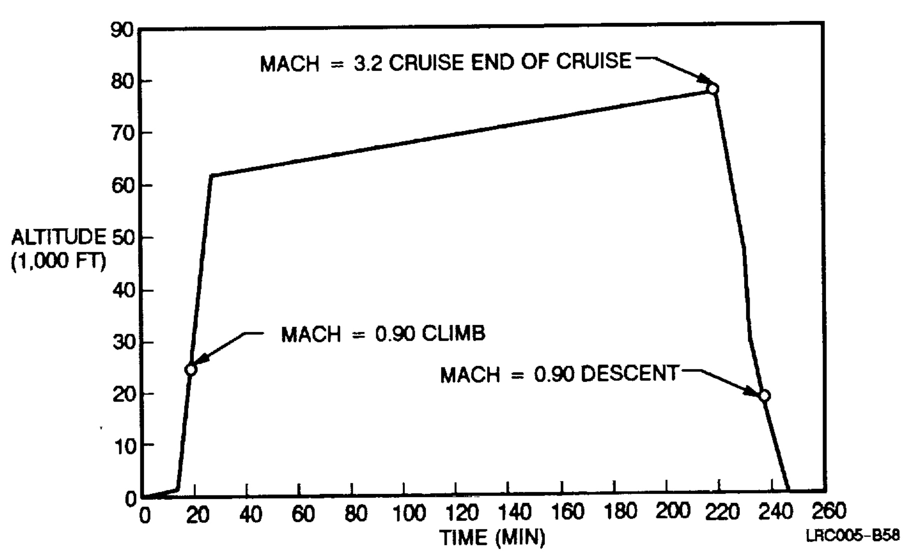


FIGURE 6-6. MACH 3.2 FLIGHT PROFILE

TABLE 6-4
DESIGN LOADING CONDITIONS

ALTITUDE (1,000 FT)	MACH NUMBER	DESCRIPTION
24.5	0.9	TRANSONIC REGION CLIMB – MAXIMUM SKIN PRESSURE
77.8	3.2	END OF CRUISE – MAXIMUM TEMPERATURE AT LOAD
17.5	0.9	TRANSONIC DESCENT – COMBINED LOADING

LRC005-B70

These conditions were selected as representative of the critical design conditions for the aircraft skin. The maximum loading case at room temperature occurs at the transonic point of Mach 0.9 climb. The maximum temperature is at the end of Mach 3.2 cruise. The third case is a combination of a high-load, low-weight aircraft at an elevated temperature. MSC/NASTRAN was used to determine the internal loads at these three loading cases. The Mach 0.9 climb at 94-percent fuel load was determined to be the critical design case for the skin structure of the aircraft.

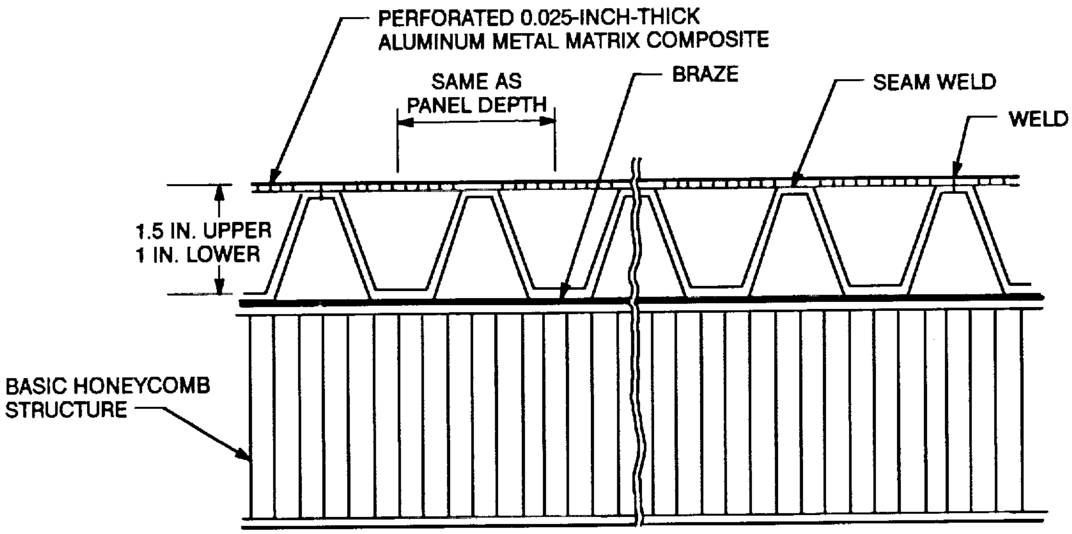
The LFC skin design is identical for both suction patterns. Two different candidate LFC skin design concepts have been developed as shown in Figure 6-7. The advantages and disadvantages of each concept are also summarized in the figure. The material selected as representative of the advanced materials properties goal is an advanced aluminum metal matrix composite (Al-MMC), which uses silicon carbide fibers and rapid solidification rate aluminum (SCS-8/RSR-Al). Since this material does not exist in a producible form, major research and development efforts will be required.

The external perforated shell of each design concept contains 0.0025-inch holes, which are distributed according to suction requirements over the surface of the shell and through which boundary layer air is sucked into the spanwise flutes and then into the pressure ducts. The spacing between these holes is 0.035 inch. The reduction in strength and fatigue/fracture resistance of the perforated aluminum metal matrix composite used in Concepts 1 and 2 is minimized because the fibers of the MMC can be in place to support the static and fatigue loads even though the matrix of the MMC laminate is perforated. The structure of Concept 1 has better damage tolerance and durability than that of Concept 2.

The basic SCS-8/RSR-Al honeycomb structure was designed using a McDonnell Aircraft Company computer program that uses in-plane loading generated by NASTRAN, normal pressure, temperatures, and combinations to perform optimum design with respect to minimum weight. The analysis and optimization that it performs are shown in Figure 6-8.

Analysis techniques for the following failure modes are included in the program:

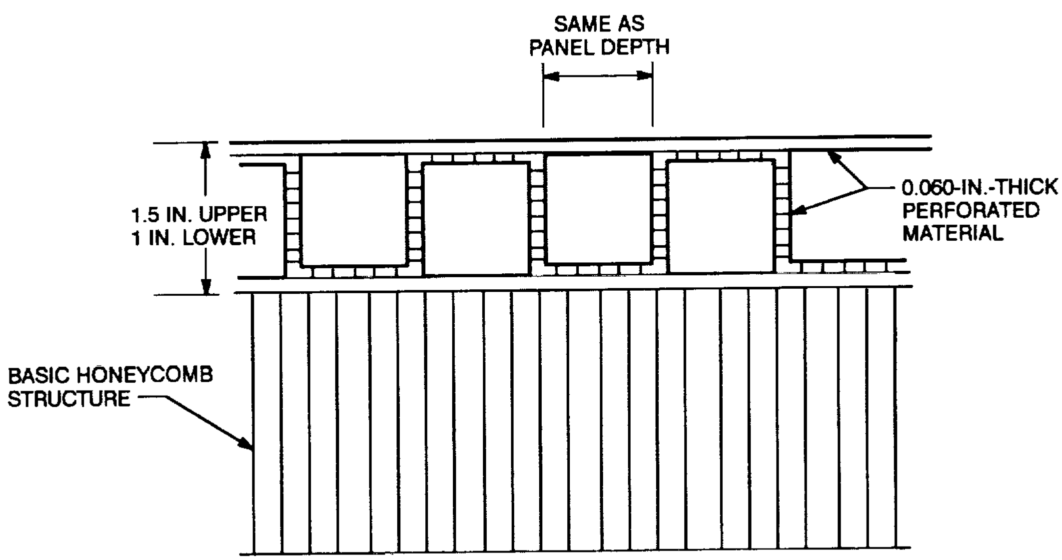
- Basic strength (elastic analysis)
- Overall panel stability (buckling)
- Local stability (buckling)
- Stiffener crippling
- Column stability
- Beam column



CONCEPT 1: ALUMINUM METAL MATRIX COMPOSITE

STRENGTH: LIGHTWEIGHT
LESS REDUCTION IN STRENGTH AND
FATIGUE AND FRACTURE RESISTANCE

WEAKNESS: NEEDS DEVELOPMENT WORK FOR HIGH MACH NUMBER APPLICATION (e.g., M = 3.2)



CONCEPT 2: ALUMINUM METAL MATRIX COMPOSITE

STRENGTH: LIGHTWEIGHT

WEAKNESS: NEEDS DEVELOPMENT WORK FOR HIGH MACH NUMBER APPLICATION (e.g., M = 3.2)
REDUCTION IN STRENGTH AND FATIGUE AND FRACTURE RESISTANCE

LRC005-B59

FIGURE 6-7. LFC STRUCTURAL CONCEPTS

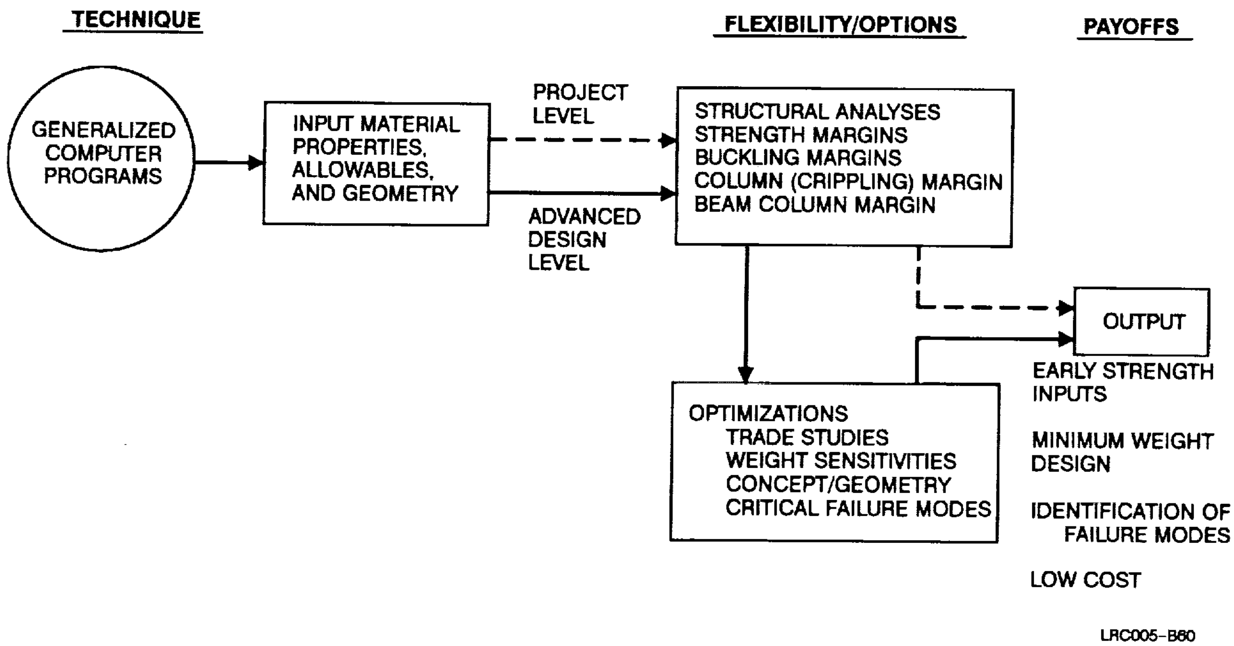


FIGURE 6-8. GENERALIZED STRUCTURAL OPTIMIZATION COMPUTER PROGRAM

All of these failure modes are addressed in each of the programs. Allowables and margins of safety are output for compression in-plane loading. For tension in-plane loading, only the basic strength checks are made, and the corresponding allowables and margins of safety are output. The basic aluminum honeycomb structures are shown in Figure 6-9. The entire aircraft is divided into six regions. The honeycomb structure consists of aluminum metal matrix outer and inner face sheets and the aluminum core. The depth of the skin (h), the unit weight of the skin (w), and the number of face sheet lay-ups in each region are defined in the figure.

Based on calculated stress requirements, the depth of the basic honeycomb structure is 1.4 inches for both the upper and lower surfaces of the inboard wing and 1.86 inches in the outboard wing region.

The integration of the LFC suction system into the wing and fuselage structure was investigated for the two suction cases. In both cases, the LFC system is subjected to low-density 500° to 600°F air, which must be ducted to compressors with the least possible reduction in wing fuel capacity. In addition, the system must allow for articulation of the leading edge flap panels.

The suction panels, previously described, that form the aerodynamic surface of the wing must meet LFC waviness and smoothness criteria based on an extrapolation of proven subsonic criteria. They have a perforated external surface through which boundary layer air is sucked into spanwise flutes integral with the panel. The flute depth and the cross-sectional area of the ducting were calculated based on the length of the flute and the suction coefficient Cq. The flute areas were integrated to determine the collector duct cross-sectional areas. The suction panel skin thicknesses were determined by the structural requirements.

For the full LFC case, the low-pressure air from the wing upper surface is ducted directly into low-pressure collector ducts on each side of the high-pressure duct from the lower surface as shown in Figure 6-10. Air collected from the upper surface panels forward of the fuel tank is ducted to the main subfloor ducts by feeder ducts that run along the sides of the cargo compartment. Air collected from the lower surface panels is ducted underneath the cargo compartment. Air from the lower surface

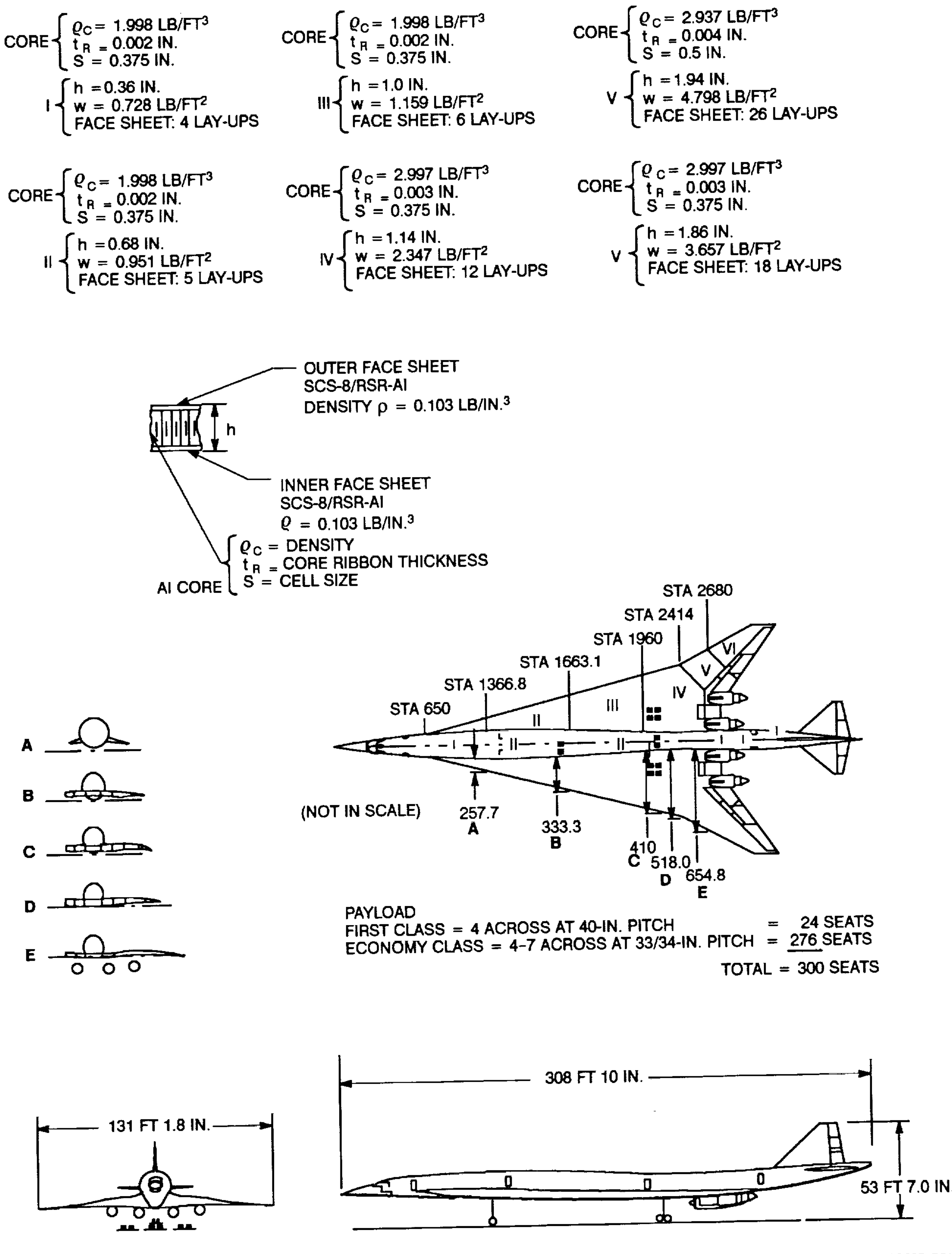


FIGURE 6-9. DESIGN OF BASIC HONEYCOMB STRUCTURE FOR MACH 3.2

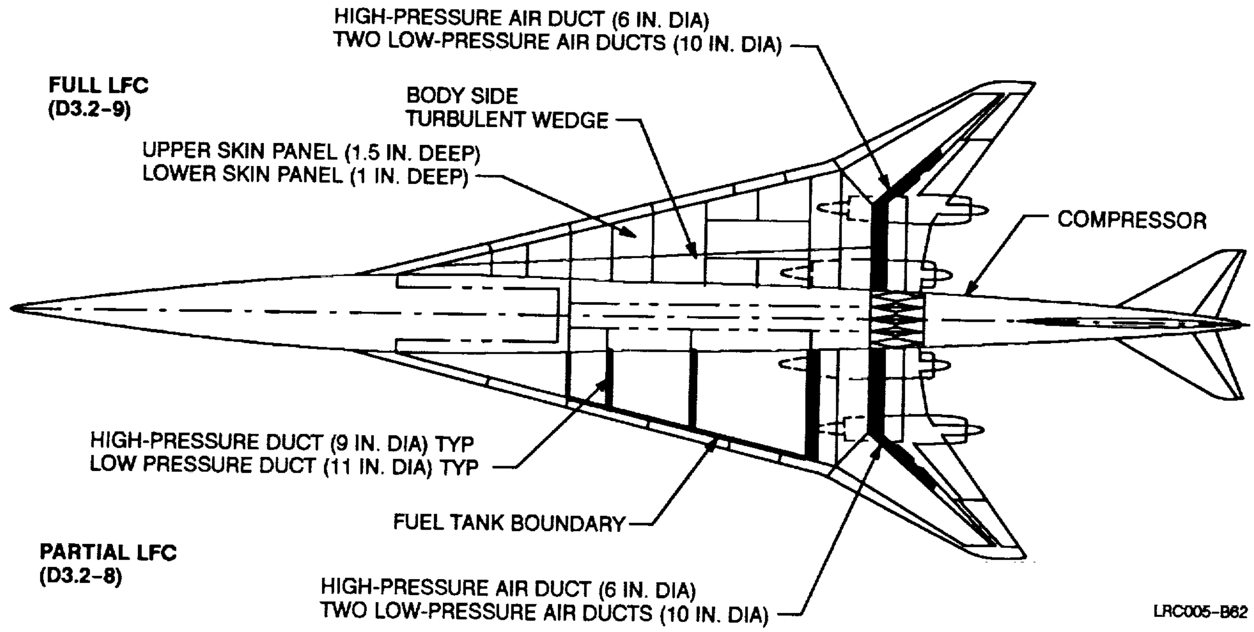


FIGURE 6-10. MACH 3.2 LFC TRANSPORT

adjacent to the landing gear door is ducted to and along the leading edge until it can be transferred to the flutes running spanwise, fore and aft of the landing gear well. The upper and lower surfaces have different suction requirements and surface flow conditions. Therefore, to minimize duct volume, suction air from either surface is handled by separate ducts and compressors.

Air collected from the wing panels over the fuel tank is ducted between the cabin floor and the wing as shown in Figure 6-11. The high-pressure air from the lower wing surface is ducted to the center duct between two ribs at the wing centerline. The fuselage ducts lead directly to the suction compressors in the aft fuselage, where the suction air is compressed and exhausted overboard. The suction air is exhausted through an optimally expanded nozzle aimed directly aft. In this manner, the ram drag penalty associated with Cq is offset by the thrust derived from the expulsion of the suction air (Reference 6-7). Integration of the compressors did not require modification of the existing area-ruled fuselage, so there was no suction system wave drag penalty.

The partial LFC system is similar to the full-chord system except that air from the leading edge panels is ducted through dry bays in the fuel tank to the fuselage as illustrated in Figure 6-12.

6.5 WEIGHTS

The weight data are based on an analysis of layout drawings of the two proposed design concepts.

The weights summarized in Table 6-5 represent LFC system weights based on availability of the material technology by the year 2000. The material selected for the purposes of this study, silicon-carbide-strengthened/rapid-solidification-rate aluminum SCS-8/RSR-Al, has a density of 0.103 lb/in^3.

An incremental weight (over non-LFC, turbulent skin panels) of 0.5518 lb/ft^2 of wetted area was determined to be required to provide the LFC skin feature (Concept 1). The outer suction surface skin is perforated by laser drilling techniques, which remove 7-percent of the material. Therefore, a 7-percent porosity weight reduction factor is reflected in the weight estimates. Except for the LFC suction panels, all other components are assumed to be identical, and weights are derived by material substitution and ratioing only.

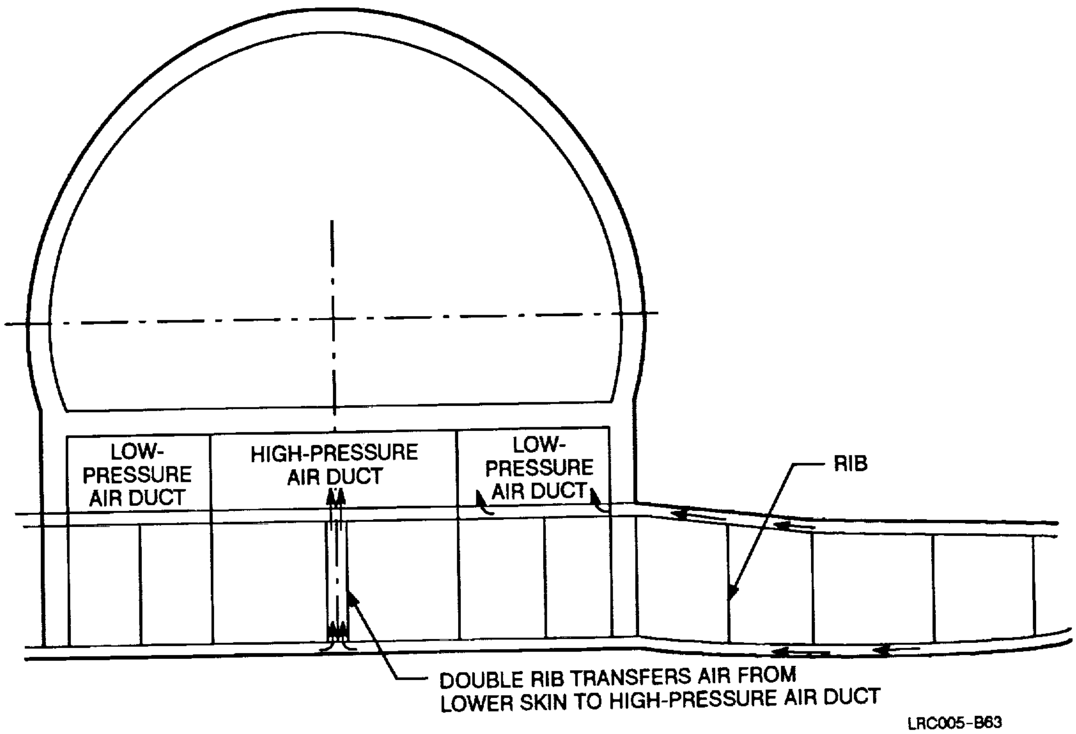


FIGURE 6-11. WING/FUSELAGE CROSS SECTION

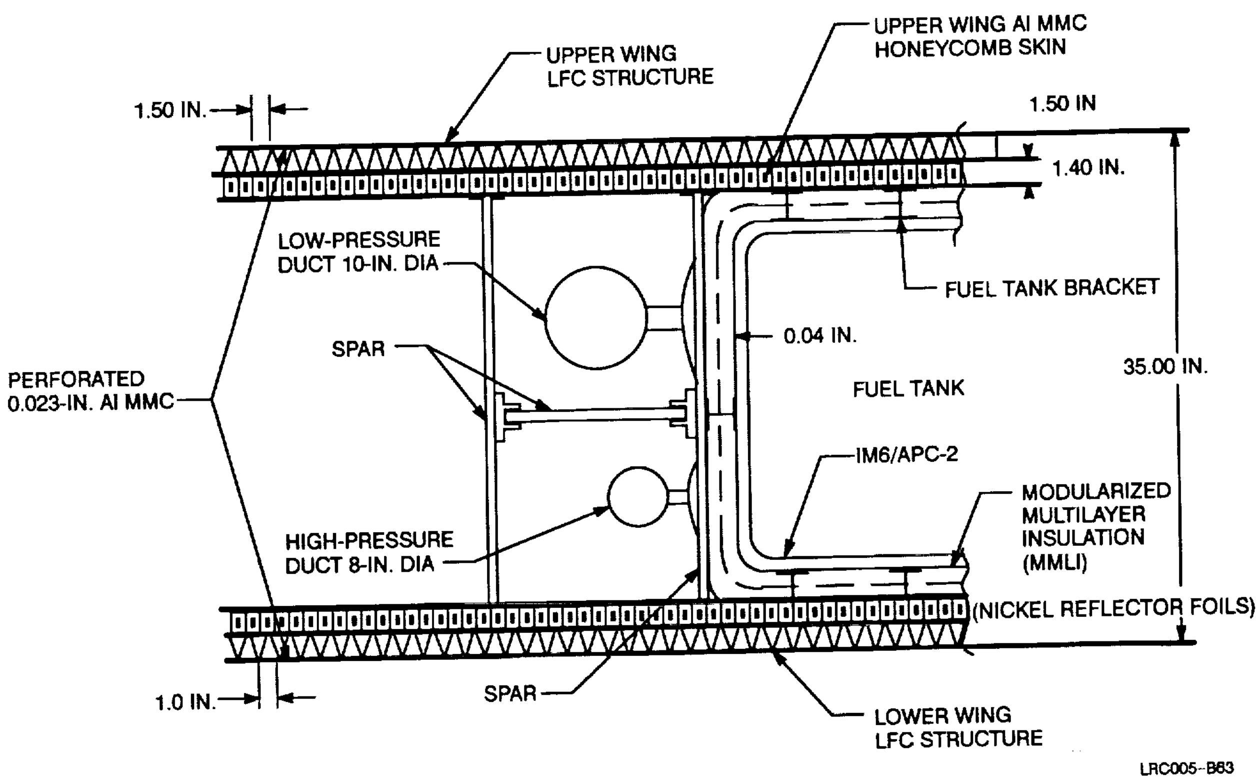


FIGURE 6-12. LFC STRUCTURAL DESIGN - PARTIAL CHORD SUCTION

TABLE 6-5
LFC SYSTEM WEIGHT SUMMARY - YEAR 2000 TECHNOLOGY

ITEM DESCRIPTION	PARTIAL LFC (LB)	FULL LFC (LB)
LFC SUCTION SURFACE COVERING	2,431	6,689
LE LFC PLENUMS, SEALS, ETC.	1,066	1,066
PLENUM MEMBRANES, CENTER FUSELAGE UNDER FLOOR	1,055	1,055
LFC PIPING AND COLLECTORS	----	252
LFC POWER PLANT INSTALLED	6,370	9,100
PIPING AND LE LFC DEDICATED STRUCTURE	1,418	----
TOTAL LFC SYSTEM WEIGHTS	12,340	18,162

YEAR 2000 TECHNOLOGY = SCS-8/RSR Al MATERIAL LRC005-B71

The weight estimates for the ducting assume that the pressurization support structure under the cabin floor will be used as part of the ducting. Skin was attached to existing vertical support structure, thus providing walls for closing off the plenums or ducts and providing the separate ducting required for the different pressure flows. The same materials and unit densities as those of the corresponding LFC panels are used in the weight estimates of the central ducting.

Most of the fuselage ducting design is shared in common for either the full or partial LFC concepts. However, there is one major difference that has a significant impact on the system weight, and that is the boundary layer bleed transfer path from the collection area to the central transfer ducting in the fuselage. In the full-chord LFC concept, the collection and transfer of the boundary layer air takes place through the fluted core sandwich of the LFC suction panels and requires no separate ducting through the wing. Ducting is required along the wing rear spar to transfer the boundary layer bleed from the outer wing to the compressors located aft of the wing in the aft body. However in the partial LFC concept, transfer of the boundary layer air from the leading edges to the central fuselage ducting would be accomplished by providing pipe ducting at several locations along the wing because there is no integral fluting in the skin panels. Separate ducting of different diameters for each respective pressure must be provided. Since these hot pipes cannot be routed through fuel bays, separate dry bays must be provided to route ducting from the wing leading edge to the fuselage central ducting. These dry bays add three extra spanwise bulkheads for each side of the wing, resulting in a significant weight penalty for this system concept.

The same number and type of LFC system components are required for either the partial or the full LFC concept. The system component weights differ because the power requirements and size are proportional to the surface area being laminarized. The weights summarized in Table 6-5 reflect those effects. The major LFC system components are located in the aft fuselage and aft of the wing. Other alternative locations were considered, but the aft fuselage location was selected for the purposes of this study.

Weights for each of the LFC concepts were incorporated into a baseline to determine the new operating empty weight (OEW). The LFC-equipped aircraft were then parametrically evaluated for mission performance and resized. The weight characteristics of the final sized partial and full LFC-equipped HSCT configurations are presented in Tables 6-6 and 6-7, respectively.

TABLE 6-6
WEIGHT DESIGN POINT SUMMARY FOR HSCT D3.2-8 – PARTIAL LFC

GEOMETRY DATA		
RANGE (N MI)	6,500	
MACH NUMBER	3.2	
NUMBER OF PASSENGERS	300	
TAKEOFF GROSS WEIGHT (LB)	799,600	
MAX ZERO FUEL WEIGHT (LB)	314,752	
MAX SPACE-LIMITED PAYLOAD (LB)	66,500	
WING AREA – THEORETICAL (FT^2)	9,500	
HORIZONTAL TAIL AREA – THEORETICAL (FT^2)	732	
VERTICAL TAIL AREA – THEORETICAL (FT^2)	669	
TURBOJET ENGINE/THRUST (SLS/ENG)	4/31,630	
SELECTED DESIGN WEIGHT DATA (LB)		**% OEW**
WING	54,719	22.04
FUSELAGE	33,344	13.43
THERMAL PROTECTION SYSTEM	11,285	4.55
HORIZONTAL TAIL	2,311	0.93
VERTICAL TAIL	2,094	0.84
LANDING GEAR	33,379	13.45
INLET, NACELLE, ENG SYS, AND MOUNT	10,531	4.24
PROPULSION SYSTEM	27,119	10.92
FUEL SYSTEM	3,790	1.53
FLIGHT CONTROLS AND GUIDANCE	9,210	3.71
FURNISHINGS	24,396	9.83
AIR-CONDITIONING	7,610	3.07
AUXILIARY POWER UNIT	0	0.00
INSTRUMENTS	1,291	0.52
HYDRAULICS	0	0.00
PNEUMATICS	0	0.00
ELECTRICAL AND LIGHTING	4,797	1.93
AVIONICS	2,480	1.00
ICE PROTECTION	343	0.14
LOAD AND HANDLING	103	0.04
LAMINAR FLOW CONTROL SYSTEM	12,340	4.97
MANUFACTURER'S EMPTY WEIGHT		241,152
OPERATOR ITEMS	7,089	2.86
OPERATIONAL EMPTY WEIGHT		248,252
MISSION PAYLOAD	61,500	
MISSION ZERO FUEL WEIGHT		309,752
MISSION FUEL	489,847	
TAKEOFF GROSS WEIGHT		799,600
FUEL FRACTION		0.613

LRC005-B72

6.6 ALTERNATE (LFC) PLANFORM

The concept of an all-supersonic leading edge wing with full-chord LFC, was addressed. This wing (D3.2-10), illustrated in Figure 6-13, was designed using the same linear theory design code that was used in the baseline design.

TABLE 6-7
WEIGHT DESIGN POINT SUMMARY FOR HSCT D3.2-9 – FULL LFC

GEOMETRY DATA		
RANGE (N MI)	6,500	
MACH NUMBER	3.2	
NUMBER OF PASSENGERS	300	
TAKEOFF GROSS WEIGHT (LB)	767,400	
MAX ZERO FUEL WEIGHT (LB)	316,401	
MAX SPACE-LIMITED PAYLOAD (LB)	66,500	
WING AREA – THEORETICAL (FT^2)	9,500	
HORIZONTAL TAIL AREA – THEORETICAL (FT^2)	732	
VERTICAL TAIL AREA – THEORETICAL (FT^2)	669	
TURBOJET ENGINE/THRUST (SLS/ENG)	4/29,800	
SELECTED DESIGN WEIGHT DATA (LB)		**% OEW**
WING	54,042	21.63
FUSELAGE	33,344	13.34
THERMAL PROTECTION SYSTEM	11,285	4.52
HORIZONTAL TAIL	2,311	0.92
VERTICAL TAIL	2,094	0.84
LANDING GEAR	32,066	12.83
INLET, NACELLE, ENG SYS, AND MOUNT	9,922	3.97
PROPULSION SYSTEM	25,550	10.22
FUEL SYSTEM	3,790	1.52
FLIGHT CONTROLS AND GUIDANCE	9,210	3.69
FURNISHINGS	24,396	9.76
AIR-CONDITIONING	7,610	3.05
AUXILIARY POWER UNIT	0	0.00
INSTRUMENTS	1,291	0.52
HYDRAULICS	0	0.00
PNEUMATICS	0	0.00
ELECTRICAL AND LIGHTING	4,797	1.92
AVIONICS	2,480	0.99
ICE PROTECTION	343	0.14
LOAD AND HANDLING	99	0.04
LAMINAR FLOW CONTROL SYSTEM	18,162	7.27
MANUFACTURER'S EMPTY WEIGHT	242,801	
OPERATOR ITEMS	7,099	2.84
OPERATIONAL EMPTY WEIGHT	249,901	
MISSION PAYLOAD	61,500	
MISSION ZERO FUEL WEIGHT	311,401	
MISSION FUEL	455,998	
TAKEOFF GROSS WEIGHT	767,400	
FUEL FRACTION	0.594	

LRC005-B73

Based on cruise performance and weight trade factors, the D3.2-10 configuration is 100,000 pounds heavier than the turbulent baseline. The full-chord LFC concept produced a 68,000-pound reduction in weight. With the entire weight benefit plus no system weight penalty applied to the D3.2-10, a 1.5-percent weight penalty exists. Therefore, the supersonic leading edge wing concept was dropped from further study.

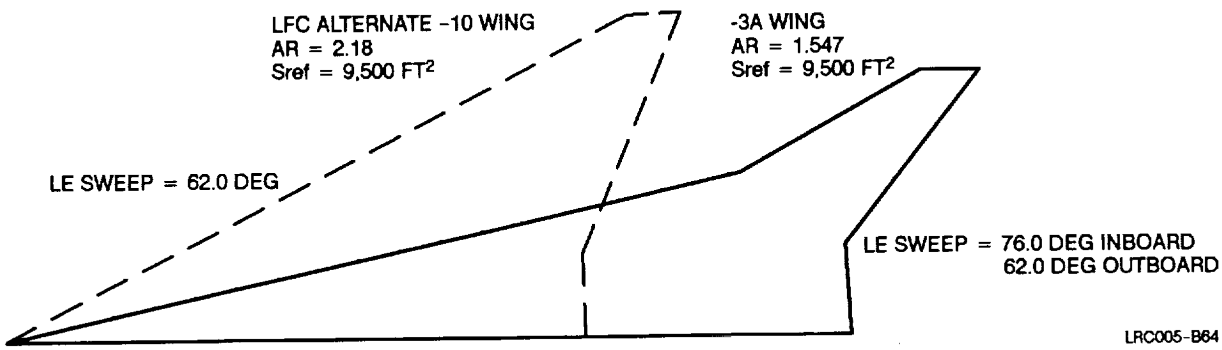


FIGURE 6-13. WING PLANFORM COMPARISON OF D3.2-3A WING AND LFC ALTERNATE WING

6.7 MISSION PERFORMANCE

Performance sizings were run on the three configurations (fully turbulent, partial LFC, and full LFC). They were sized using the Pratt & Whitney STF905 variable-stream-control engine from Phase III for a 6,500-nautical-mile range and a takeoff field length (TOFL) of 10,600 feet for an ISA day. The results, indicating a 7-percent reduction in block fuel for the full LFC over the partial LFC concept, are shown in Table 6-8. The power extracted to drive the LFC pumping system is less than 1 percent of the total power generated by the propulsion system during cruise and was therefore neglected in this analysis.

TABLE 6-8
HSCT D3.2-7/D3.2-8/D3.2-9 SIZE AND PERFORMANCE COMPARISON

	FULLY TURBULENT D3.2-7	PARTIAL LFC D3.2-8	FULL LFC D3.2-9
TOGW (LB)	835,500	799,500	767,384
Fn (LB)	70,323	65,955	62,139
OEW (LB)	240,673	248,025	249,674
BLOCK FUEL (LB)	477,920	438,053	408,803
L/D AVG	8.40	8.97	9.35

TOFL = 10,600 FT, RANGE = 6,500 N MI
SL STANDARD DAY
S_W = 9,500 FT^2
LRC005-B74

The effect on range of a loss of LFC is shown in Figure 6-14. For the design range of 6,500 nautical miles from Los Angeles to Sydney, the effect of the worst case, full LFC failure on the D3.2-9, was studied. Loss of LFC is not critical for every point along the flight path; there is always a suitable runway available. However, if LFC is lost beyond 5,600 nautical miles, reserve fuel for flight to an alternative field is needed to reach Sydney.

6.8 ECONOMIC ASSESSMENT

Aircraft worth is the investment value of an airplane to the airline operator. The worth of an HSCT to the operator includes a target rate of return on investment to the operator. This includes 1987 tax law and depreciation schedules, life of the asset and, most importantly, the annual operating cash flow. All of the airplane characteristics such as size, weight, speed, lift-to-drag ratio, propulsion efficiency, and other parameters are embodied in the cash flow estimates. Also involved in aircraft worth

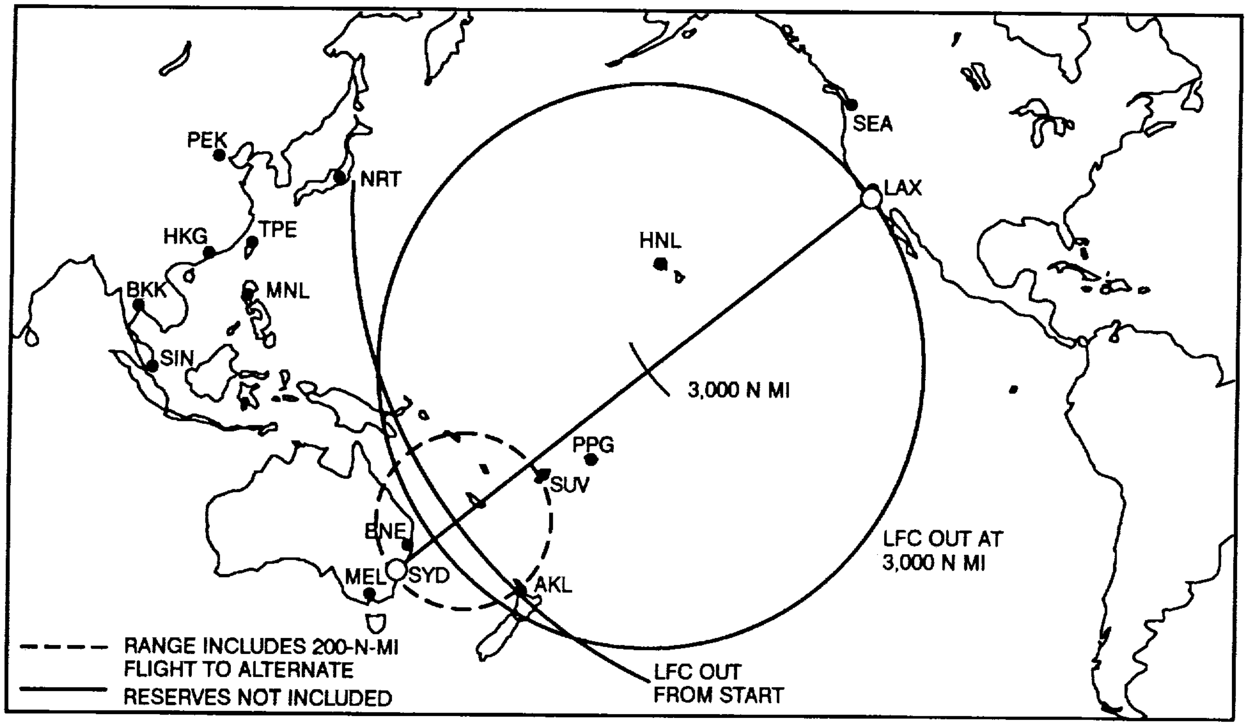


FIGURE 6-14. RANGE WITH LOSS OF LFC, FULL LFC D3.2-9

are operational parameters such as utilization, turnaround time, passenger mix, load factor, passenger fares for various regions of the world, and fare premium (for sensitivity analysis).

Cost/prices were developed using the total McDonnell Douglas base of experience and knowledge in the field of high-speed technology and support efforts (e.g., materials and processes). This base provided the benchmarks for the estimating process. Labor and material resources were estimated on a discrete evaluation basis coupled to the analogous technique. Resources were estimated by major aircraft system/component and by functional category. Development costs included all of the necessary resources and tasks required to design, develop, produce, and demonstrate an aircraft that can be FAA certified.

Labor hours were translated into 1987 dollars using the aerospace fully burdened labor rates for the different categories of labor (e.g., engineering, tooling, quality assurance, etc.). Material and equipment were estimated separately, except that propulsion system costs were furnished by the engine companies.

The end product of the estimating process is a flyaway cost (price) in which the development cost is amortized over each assumed production program with a manufacturer's targeted rate of return.

A necessary condition for program viability is to determine whether there is a feasible operational plan and ticket pricing policy so that (1) airplane revenues will cover operating costs plus an attractive rate of return to the operator and (2) fares are low enough to provide an HSCT market large enough (i.e., production volume) to permit a selling price equal to or lower than the investment value of the airplane. The evaluation procedure places the HSCT in competition against the advanced long-range subsonic airplane on a city-pair basis. This ensures that the HSCT is applied to those markets in which it best performs. Repetition of this procedure for various fare levels will determine whether

the above conditions can be simultaneously satisfied. Flyaway prices and vehicle worth are shown in Figure 6-15 as a function of quantity for all configurations. Results shown in Table 6-9 and Figure 6-16 indicate the greatest potential for net economic benefit, or the existence of a positive return on investment (ROI), lies with the full-chord LFC concept. Table 6-9 shows a 50-percent increase in fuel price does not alter the conclusion that the full-chord LFC system represents the best configuration economically.

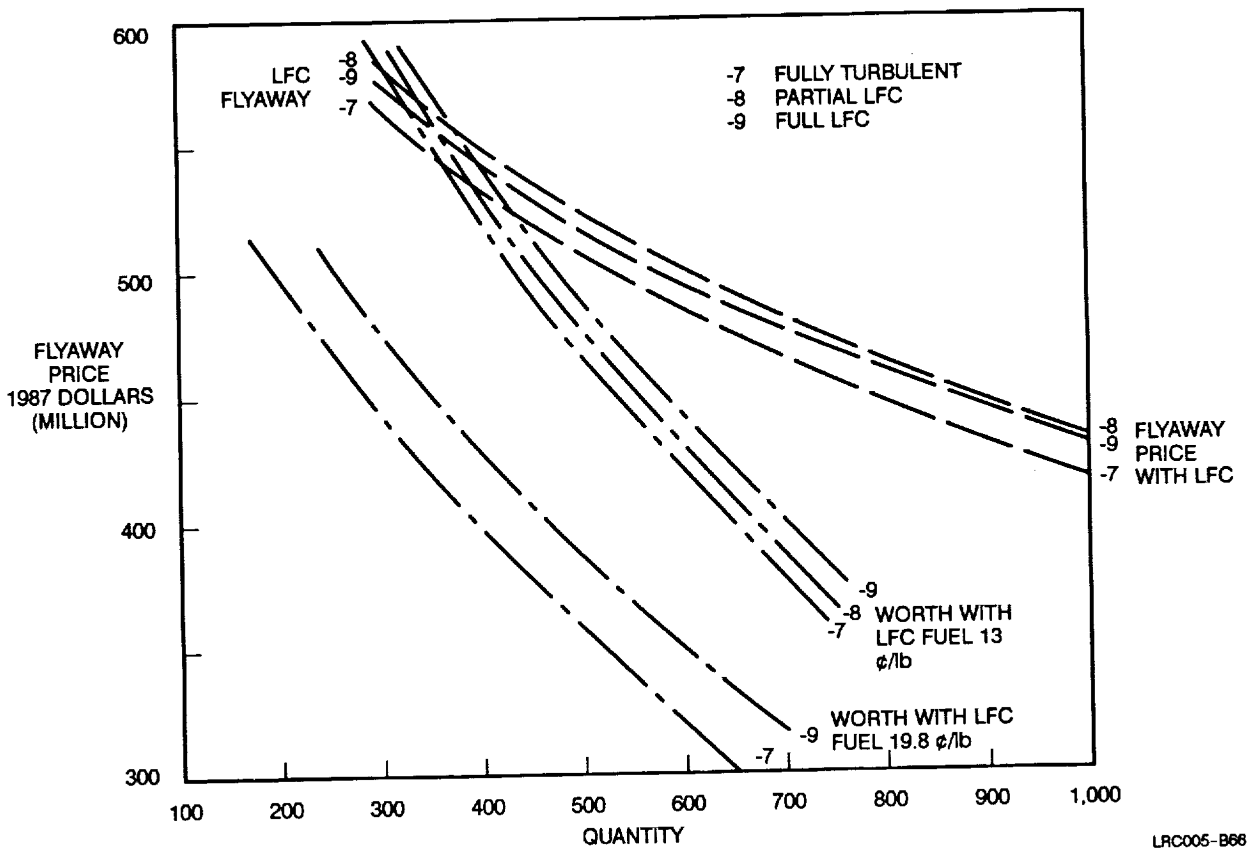


FIGURE 6-15. FLYAWAY PRICE VERSUS QUANTITY FOR GIVEN CONFIGURATIONS

TABLE 6-9
LFC ECONOMIC ASSESSMENT
350 UNITS ($ MILLION)

	FUEL PRICE = 13.2 ¢/LB			19.8 ¢/LB (+50%)		
	D3.2-7	-8	-9	D3.2-7	-8	-9
VEHICLE WORTH	548	560	570	420	---	450
FLYAWAY PRICE	548	564	556	548	---	556
NET	0	-4	+14	-128	---	-106

NET = (VEHICLE WORTH) - (FLYAWAY PRICE)

LRC005-B75

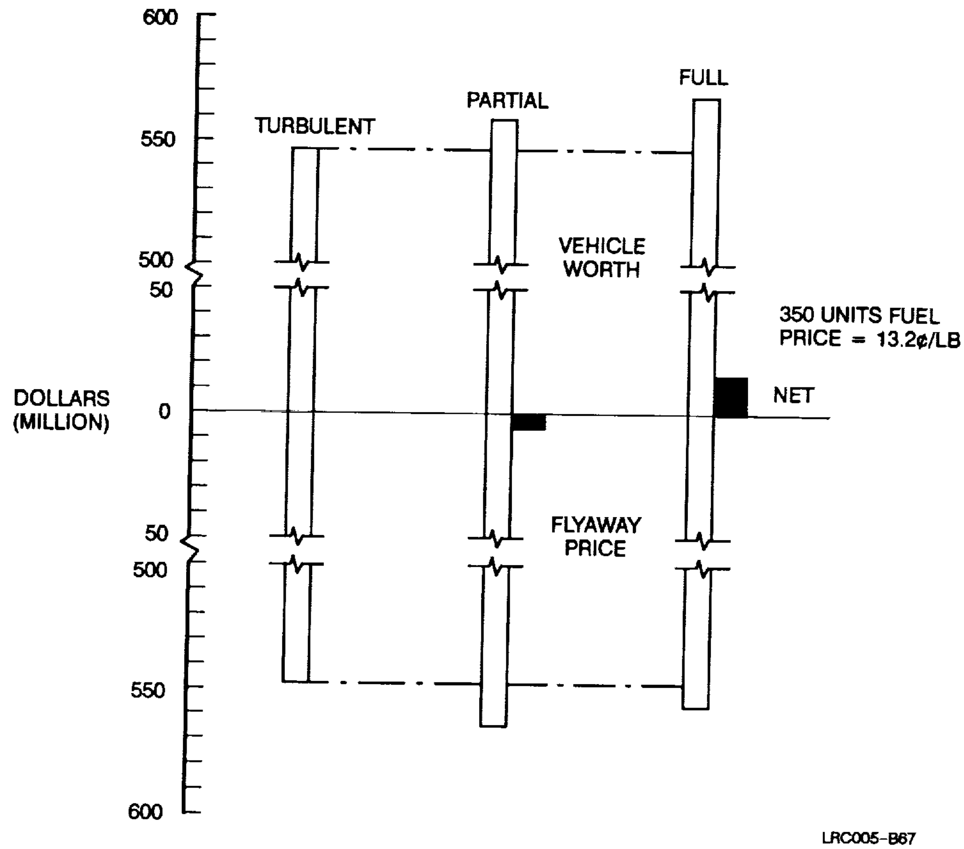


FIGURE 6-16. ECONOMIC ASSESSMENT OF LFC

6.9 CONCLUSIONS

The full-chord LFC concept resulted in a 4-percent drag reduction relative to the partial LFC concept. Structurally, it represented a simpler system because no dry bays in the fuel tank were necessary since the suction flutes were utilized to transfer suction air from the leading edge to the ducts in the fuselage. The configuration also required 7 percent less fuel (block fuel reduction) than the partial LFC concept. Thus, from an engineering standpoint, the full-chord LFC approach would appear to be the best concept to pursue. The economic evaluation of the three configurations also identified the greater value of the full-chord system in terms of a greater return on investment (ROI) for the airline, over the baseline. Therefore the full-chord LFC concept is recommended as the best option.

6.10 RECOMMENDATIONS

The LFC design study has highlighted some design issues which should be addressed in order to achieve technology readiness. In terms of enhancing the existing design and achieving design closure, it is recommended that the following tasks be performed:

- Assessment of a weight and volume penalty for additional cabin thermal protection required for the suction air ducting under the cabin floor.
- A comprehensive aircraft rebalancing for both the partial and full LFC configurations.
- A detailed analysis of the economic impact of LFC system maintenance costs on the computed aircraft worth.
- A competitive analysis of suction skin structural design concepts using various materials.

- A compressor design trade study of direct shaft versus electric motor-driven configurations.
- A study of landing gear relocation to simplify suction air flute and duct routing.
- A trade study of wing- versus fuselage-mounted compressors.
- A concept definition of the LFC systems integrated with structural requirements.
- Coupon tests of perforated skin to demonstrate fatigue and fracture characteristics.
- Assessment of manufacturability of advanced LFC skin materials.
- A study quantifying the effect of resizing for LFC on environmental factors, e.g., airport noise, sonic boom, and emissions.

6.11 REFERENCES

6-1 Jameson, A., "A Vertex-Based Multigrid Algorithm for Three-Dimensional Compressible Flow Calculation," presented at the ASME Symposium on Numerical Methods for Compressible Flow, Anaheim, California, December, 1986.

6-2 Fischer, M.C., "Spreading of a Turbulent Disturbance," AIAA Journal, Vol. 10, No 7, pp. 957-959, July 1972.

6-3 Dagenhart, J.R., "Amplified Crossflow Disturbances in the Laminar Boundary Layer on Swept Wings with Suction," NASA TP-1902, November 1981.

6-4 Malik, M.R., "COSAL — A Black Box Compressible Stability Analysis Code for Transition Prediction in Three-Dimensional Boundary Layers," NASA CR-165925, May 1982.

6-5 Kaups, K, and Cebeci, T., "Compressible Boundary Layers with Suction on Swept and Tapered Wings," J. Aircraft, Vol. 14, 1977.

6-6 Gentry, A.E.; Smyth, D.N.; and Oliver, W.R., "The Mark IV Supersonic-Hypersonic Arbitrary-Body Program," Volume 1 — Users Manual, AFFDL-TR-73-159 (available DTIC as AD778443) U.S. Air Force, November 1973.

6-7 Powell, A. G.; Agrawal, S; and Lacey, T. R., "Feasibility and Benefits of Laminar Flow Control on Supersonic Cruise Airplanes," NASA CR-181817, July 1989.

6-8 Part 25, Airworthiness Standards: Transport Category Airplanes, Federal Aviation Regulations, Federal Aviation Administration, Department of Transportation 1974.

SECTION 7
CONCLUSIONS

The following is concluded from the studies conducted in the environmental areas of sonic boom, exterior noise, and engine emissions:

- Sonic Boom — Neither aircraft operational techniques nor minimization of N-wave type sonic boom signatures will be sufficient to achieve acceptable sonic boom loudness levels. Novel concepts such as vehicle shaping with significant lift carried forward are required to lower sonic boom levels for the Mach 3.2 configuration. The lowest level obtained in this study by waveform shaping was a perceived loudness of 96.5 PLdB.
- Exterior Noise — Of the six engine cycles considered in this study, none achieved both the aircraft range and noise goals.

  GE None of the GE engines met the noise goal. The GE Study A2 three-stream and Study A1 two-stream high-flow fan engines were the only engines to achieve the 6,500-nautical-mile range goal at reasonable takeoff weights, but were 7 to 9 EPNdB above the Stage 3 sideline noise limits. An exhaust noise suppressor will be required for these engines.

  P&W None of the P&W engines, which included a high specific thrust cycle with a noise-suppression system and an engine with a high-flowing exhaust nozzle, met the aircraft range goal. However, the P&W VSCE with a mixer/ejector did achieve Stage 3 goals at a range of approximately 5,500 nautical mile.
- Emissions — For the P&W VSCE, the rich-burn/quick-quench combustor combined with a premixed pilot stage in a conventional duct burner shows promise of significant reduction of NO_x levels. This combustor technology has reduced risk levels relative to concepts studied previously according to engine company determinations.
- Laminar Flow Control — With regard to the potential for gross weight reduction through laminar flow control technology, the full-chord LFC concept proved to be preferred over fully turbulent and partial LFC concepts from both engineering and economic considerations. LFC also offers sonic boom, engine emissions, and exterior noise advantages by virtue of lower gross takeoff and cruise weights.

SECTION 8
RECOMMENDATIONS

Based on the activities summarized in this report, it is recommended that the following technology developments be conducted to continue the significant progress accomplished in Phase IIIA:

- Sonic Boom — Continue the sonic boom waveform shaping studies, concentrating on the vehicle integration and flying qualities of the aircraft. Lower cruise speed characteristics and the development and implementation of higher order methodologies applied to unique planform shapes and engine exhaust simulation must be emphasized. The prospect of minimizing annoyance at Mach numbers less than 3.2 should be investigated in combination with the current Mach 3.2 cruise. Human response studies to determine acceptable boom metrics and levels must continue to establish timely design requirements.
- Exterior Noise — The P&W mixer/ejector noise reduction concept should be studied for both the VSCE and TBE, and weight and noise reduction characteristics should be established by analysis and test. Studies of alternative GE and P&W high-flow engine cycles incorporating a suppressor should continue. Operational procedures and high-lift devices to minimize community noise using these advanced engines and suppression devices should be incorporated in future studies.
- Engine Emissions — Total annual fleet fuel burn emission scenarios and atmospheric modeling techniques to determine ozone impact should be emphasized during further studies. The development of low-emission combustor technology should continue and simultaneous trade studies should be conducted assessing engine emission and aircraft performance to minimize total emissions per flight and reduce risk.
- The LFC integration studies should be continued to validate in more detail the results achieved in Phase IIIA. Selection of the appropriate suction compressors and ducting requires more study.
- Small-scale coupon testing of various aircraft structural materials should be conducted to establish a data base appropriate for high-temperature porous surfaces required for the HSCT. Several innovative structural design concepts for the vehicle should be identified and evaluated to establish the minimum weight and maintenance combination.
- Low-speed high-lift devices will be essential to reduce community noise under the takeoff and approach flight paths. Innovative low-speed concepts with high L/D should be identified; low-speed wind tunnel tests should be conducted for promising high-lift devices.
- Studies to reduce fuselage turbulent drag should be initiated. These need to include aircraft wing resizing for maximum reduction in fuel and weight.
- Detailed economic trade studies should be conducted to cover the environmental technology areas affecting sonic boom, engine emissions and exterior noise.

APPENDIX

CONCEPT DESCRIPTION

A detailed description of the Phase III baseline configuration D3.2-3A (Figure A-1) and its characteristics are provided in Reference 2-2. A summary description is presented in this appendix. The concept features a double-sweep arrow planform wing, conical-taper single-lobe fuselage, aft vertical and horizontal empennage, four Pratt & Whitney STF947 variable-stream-control engines, and a tricycle landing gear. The fuselage was designed to accommodate a nominal seating arrangement of three classes: 10, 30, and 60 percent for first, business, and coach classes, respectively. Supersonic drag considerations necessitated use of a varying cabin cross section. The fuselage incorporates single-lobe shaped cross sections with the width varying according to longitudinal location. The maximum section is determined by twin aisles with seven-across coach seats; the minimum section is determined by a single aisle with a five-across seating arrangement. The maximum section will accommodate six-across first/business-class seats, and the minimum section will hold four-across business-class seats. All seat sizes are consistent with those used on MD-80 and MD-11 aircraft.

The baseline interior arrangement (Figure A-2) provides contemporary service for 300 passengers based on a maximum flight duration of 5 hours. Each class (first, business, and coach) has its own galley, lavatories, coatrooms, and cabin attendant stations. Four cabin doors per side are evenly distributed for rapid evacuation. Adjacent flight attendant seats are provided. Slide packs are located on each door. Cabin windows are incorporated. Five operational interior arrangements have been defined reflecting two- and one-class seating options. Capacity varies from 239 seats in an all-business configuration to 392 seats in an all-coach configuration.

AERODYNAMICS

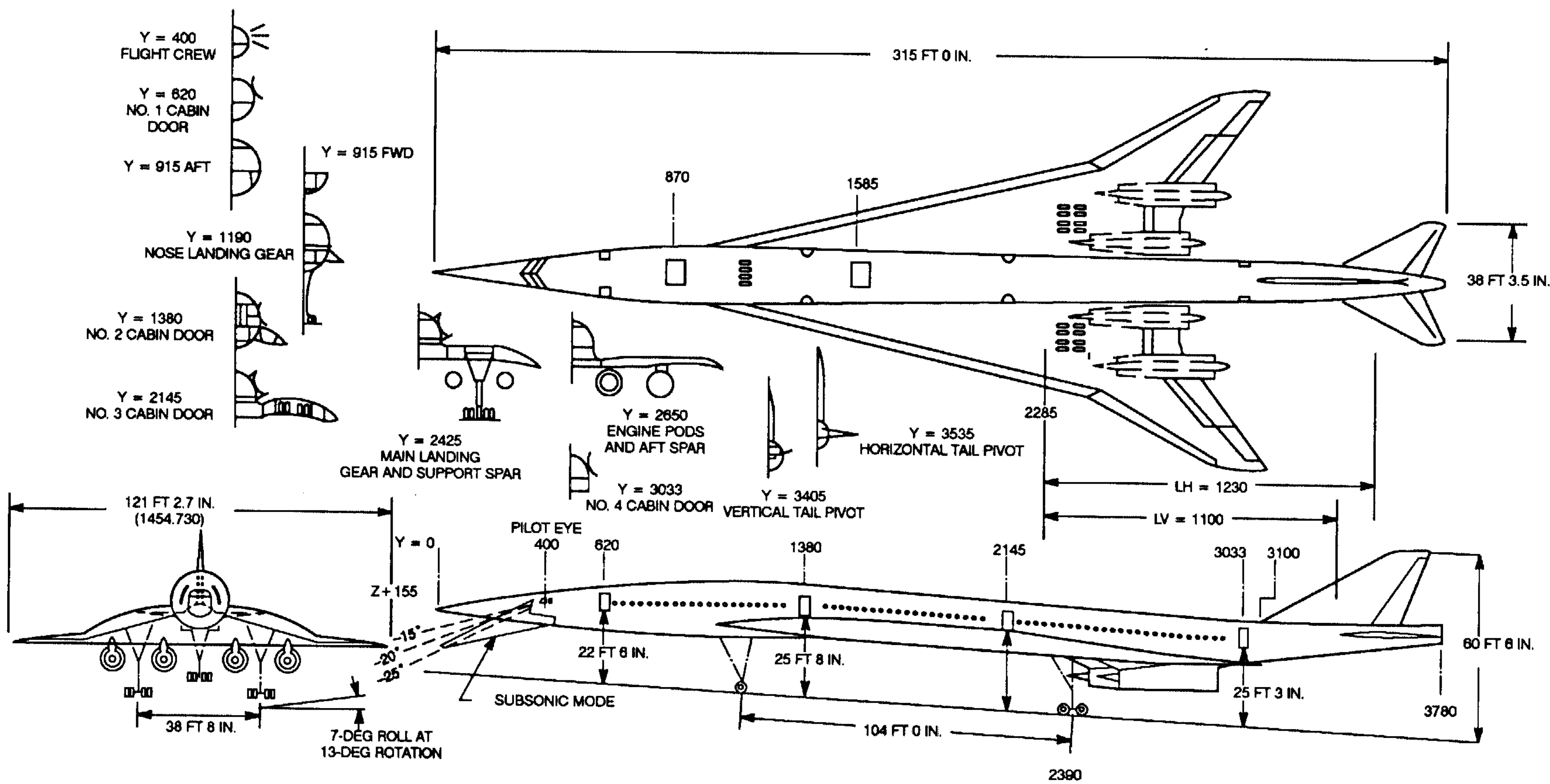
The D3.2-3A concept was based on the preceding Advanced Supersonic Transport (AST) developed under joint NASA/McDonnell Douglas funding. The basic arrow-wing AST planform was modified with increased leading- and trailing-edge sweep to improve supersonic performance at Mach 3.2 cruise. The wing design had a planform reference area of 9,500 square feet, an aspect ratio of 1.547, inboard leading edge sweep of 76 degrees, sweep break at 65-percent semispan, and outer panel leading edge sweep of 62 degrees. (See Table A-1.)

The wing camber was optimized for a maximum wing-body trimmed lift-to-drag ratio at cruise for $C_L = 0.091$. Wing thickness distribution was based on previous AST studies. The wing airfoil is a modified NACA 64 series airfoil inboard of the planform break and biconvex section outboard of the planform break. The fuselage area distribution and camber were optimized for a minimum wave drag due to volume at Mach 3.2 cruise conditions. The nacelles were staggered to minimize wave drag. Laminar flow control (LFC) was included in the D3.2-3A baseline concept and aerodynamic analysis. Inboard of the planform break, LFC suction region was limited by the fuel tank boundaries. Outboard of the planform break, suction was applied to the flap hinge line. The cruise lift/drag characteristics are shown in Figure A-3. The high-lift system consisted of plain trailing-edge flaps and full-span single-drooped leading-edge flaps (Figure A-4).

Control surfaces required for D3.2-3A are illustrated in Figure A-5. Longitudinal control and trim capability were provided by a movable horizontal surface with four separate geared elevators. Ailerons and multiple spoiler panels provided lateral control. Directional control was provided by a rudder. Stability and control augmentation was required on all axes. On the longitudinal axis, negative stability margin of 10 percent was assumed, which requires fuel management and pitch-up compensation systems.

PAYLOAD

FIRST CLASS	=	10 PERCENT	=	4 ACROSS AT 42-IN. PITCH	=	30 SEATS
BUSINESS CLASS	=	30 PERCENT	=	4-6 ACROSS AT 38-IN. PITCH	=	90 SEATS
COACH CLASS	=	60 PERCENT	=	4-5 ACROSS AT 32-IN. PITCH	=	180 SEATS
TOTAL	=	100 PERCENT			=	300 SEATS



LRC005-78

FIGURE A-1. GENERAL ARRANGEMENT - D3.2-3A CONCEPT

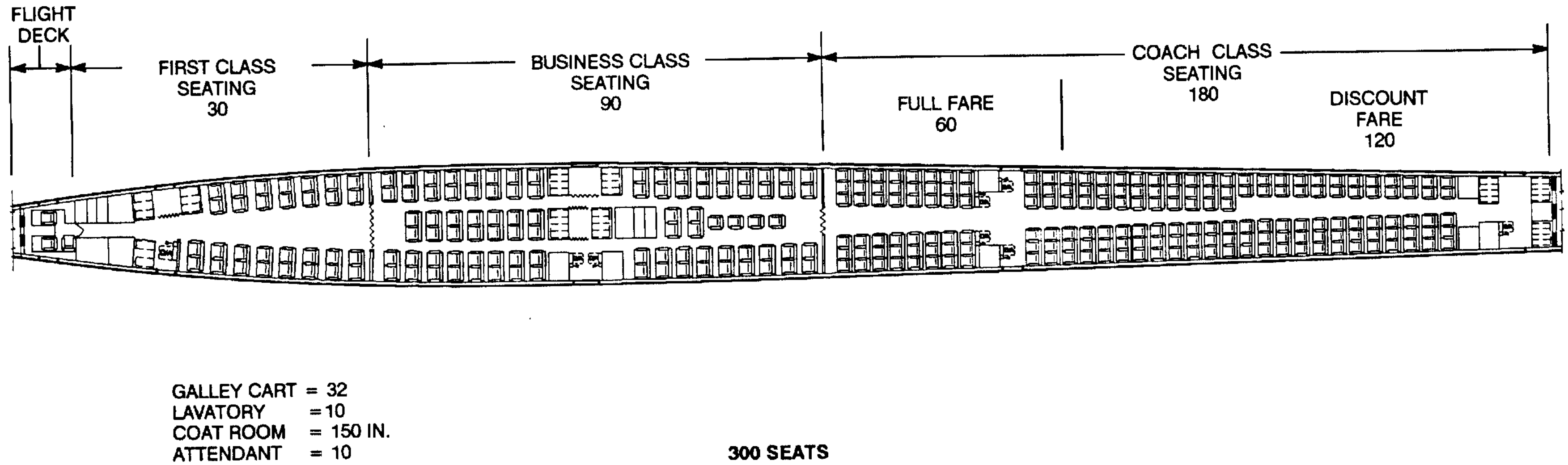


FIGURE A-2. D3.2-3A CONCEPT BASELINE INTERIOR

TABLE A-1
D3.2-3A CONCEPT WING PLANFORM SUMMARY

CONFIGURATION	D-3.2-3A
REFERENCE AREA (FT^2)	9,500
ASPECT RATIO	1.547
LEADING EDGE SWEEP (DEG)	76/62
SPAN BREAK (%)	0.65
HIGH-LIFT DEVICES	LEADING EDGE FLAPS TRAILING EDGE PLAIN FLAPS

LRC005-B77

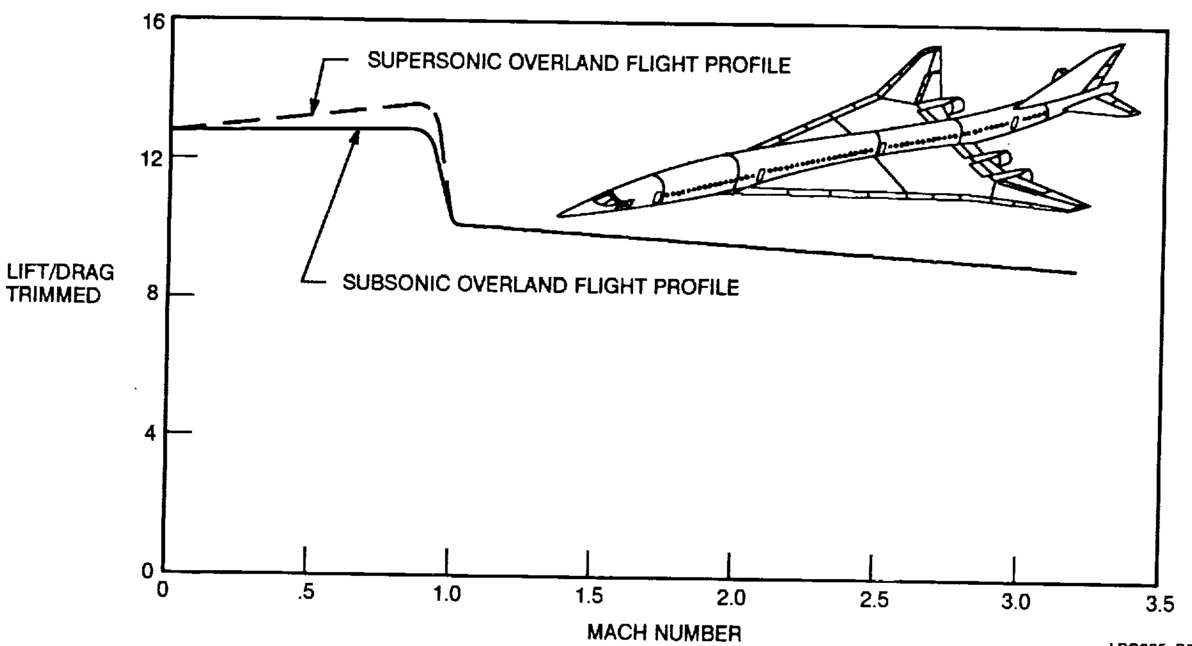
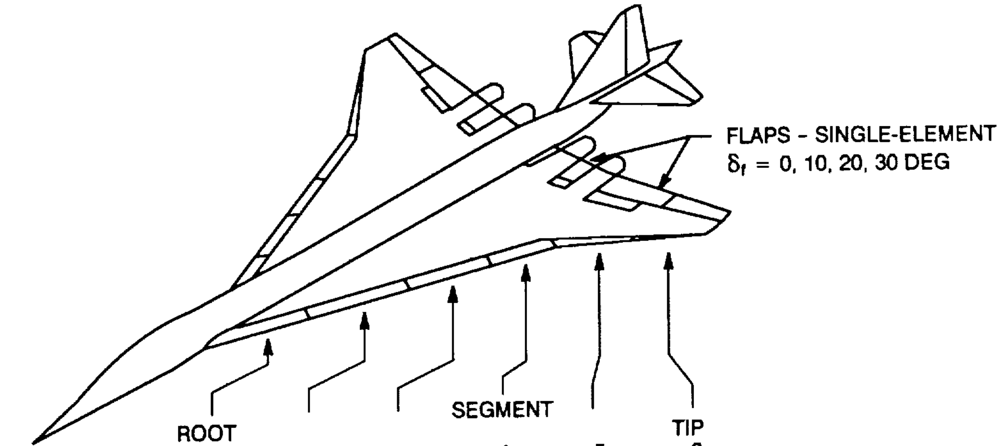


FIGURE A-3. LIFT/DRAG RATIO FOR MACH 3.2 CONFIGURATION

PROPULSION

The Mach 3.2 baseline engine is the Pratt & Whitney variable-stream-control engine (VSCE) duct-burning nonmixed-flow turbofan using thermally stable jet fuel (TSJF). The unscaled engine has a design-corrected airflow of 650 pounds per second and maximum augmented and dry SLS thrust ratings of 61,901 and 29,694 pounds, respectively.

The VSCE (Figure A-6) is an advanced, moderate-bypass-ratio nonmixed-flow turbofan with duct burner augmentation and a coannular nozzle with inverted velocity profile for jet noise reduction. At takeoff, the main burner is throttled to an intermediate power setting to reduce the core contribution to jet noise. The duct burner is operated at a moderate temperature level to provide the required thrust. During subsonic cruise, the VSCE operates as a moderate-bypass turbofan engine. The main burner operates at a relatively low exit temperature, with no duct augmentation. During supersonic cruise, the main burner temperature high spool speed is increased to maintain high-flow conditions.



SCHEDULE	ROOT 1	2	3	SEGMENT 4	5	TIP 6
CLEAN	0	0	0	0	0	0
SCHEDULE "R" (DEG)	13	34	35	35	19	29

LEADING EDGE FLAP DEFLECTION SCHEDULES

LRC005-B80

FIGURE A-4. HIGH-LIFT SYSTEM FOR D3.2-3A

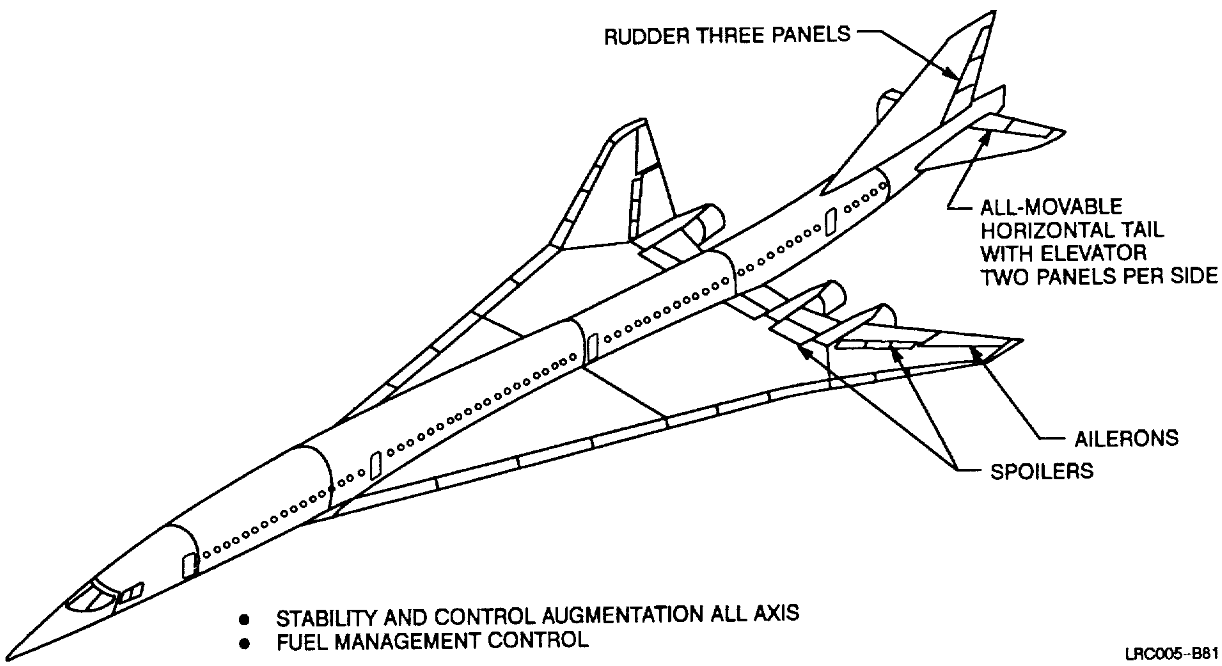


FIGURE A-5. D3.2-3A CONCEPT CONTROL SYSTEM DESIGN FEATURES

A variable-geometry bicone inlet was selected, and the inlet system designed for Mach 3.2 cruise at an altitude of 70,000 feet. The nozzle concept incorporates a combination of noise-suppression devices to meet FAR Part 36, Stage 3 noise limits. Inlet bleed air was used for nozzle/engine cooling and injected into the engine exhaust for noise reduction. The suppressor nozzle has a fixed duct stream jet area when deployed for sideline and community noise reduction.

The engines are individually mounted in nacelles located on the aft section of the wing. Wing-mounted pylons support the nacelles. A schematic of the nacelle, including the inlet, engine, and nozzle installation is shown in Figure A-7.

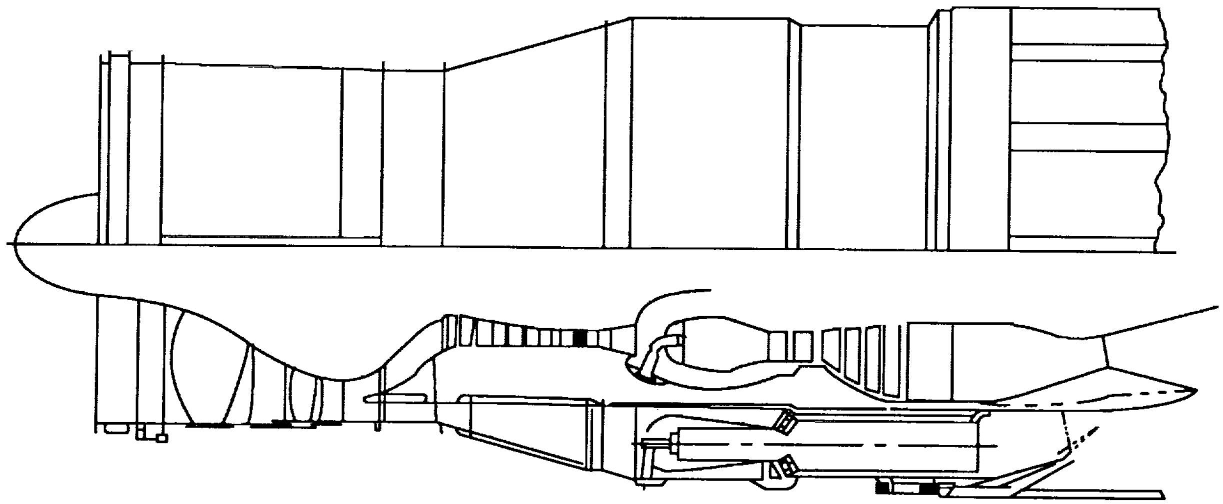


FIGURE A-6. P&W MACH 3.2 VSCE DUCT BURNING TURBOFAN

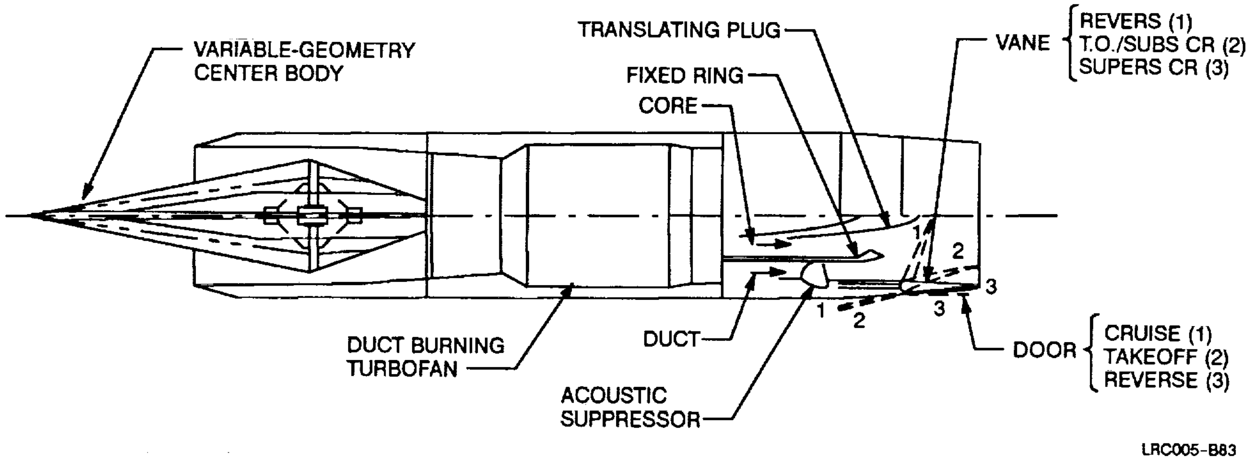


FIGURE A-7. MACH 3.2 NACELLE FEATURES, P&W VSCE DUCT BURNING TURBOFAN

STRUCTURES

The airframe structural concept consists of honeycomb load-carrying skins for the outer fuselage shell, wing, and empennage. Substructure consists of frames, ribs, and spars. Honeycomb skins carry all the body bending loads and internal pressure of the passenger cabin for safe-life design conditions. The fuel tanks only react to internal loads.

The structural definition resulted from strength analysis of combined force and thermal loads for critical loading conditions. Buckling, crippling, stiffness, tension, and other failure modes were assessed in the selection of materials for the primary structure of an aircraft. Strengthening the structure to withstand buckling and crippling accounts for approximately 75 percent of the primary structural weight. SCS-8/RSR Al (AlMMC) is the most efficient material from the buckling-crippling standpoint even though the rating of tension and stiffness falls below titanium metal matrix composite. Thus, AlMMC was the primary structural material used for the Mach 3.2 concept.

SYSTEMS

Advanced-technology subsystem improvements, considered attainable by year 2000, have been utilized in the subsystem architecture for this study.

The thermal protection system (TPS) requirements were established from a thermal management analysis. Because of external skin temperatures on the order of 600° to 700°F caused by aerodynamic heating, a passive cooling TPS for the passenger cabin and fuel tanks is required. Modularized multilayer insulation (MMLI) with wire mesh separators is utilized for its high resistance to thermal radiation and low weight. The Mach 3.2 engine nozzle uses a passive low-Q TPS.

The landing gear is a tricycle gear arrangement The main gear consists of two shock struts. Each strut includes a truck beam, eight wheels, and eight carbon brake assemblies. The nose gear consists of a single shock strut that includes a four-wheel assembly.

The engine systems include the accessory drive system, start system, lube system, cooling and ventilation system, ignition system, and engine control.

The thermally stable jet fuel system is based on a wet wing design. The system is configured for fuel management with separate nonintegral wing tanks to maintain the aircraft center-of-gravity within prescribed limits. The system includes sealant, pumps and controls, fill, drain, transfer and crossfeed, vent, and distribution systems.

The flight controls and flight guidance system are all-electric with fiber-optic signaling. The flight controls system includes the cockpit, leading and trailing edge flaps, spoilers, outboard aileron, movable horizontal stabilizer and elevators, and rudder. The controls are powered by integrated actuator packages (IAP).

The flight guidance system is an advanced integrated computational complex that includes dedicated digital computers, air data, data communication, and maintenance assessment functions, and also facilitates fly-by-light and flight controls.

Engine bleed air, which replenishes the cabin air, is cooled by vapor cycle units and a heat transfer loop system. The air-conditioning and pressurization system includes heat exchangers, electrically powered vapor cycle refrigeration units, cold air fans, engine bleed air ducting for cabin air supply, fuel-to-air heat transfer system, conditioned cabin air ducting, interwall recirculation, avionics compartment cooling, radar compartment cooling, cargo compartment cooling, lavatory and galley cooling, and cabin pressurization control system.

The electrical power and lighting system includes advanced engine-driven high-voltage direct current generators and cooling equipment, advanced solid-state control circuitry, lightweight power cables and distribution wiring, emergency battery and charger system, and static inverter system. The lighting system incorporates external and internal lights, light fixtures, and circuitry.

The ice protection system includes an all-electric forward-vision windscreen with anti-ice, defogging, and rain removal system, and anti-icing for the engine inlet leading edge and centerbody.

The laminar flow control system includes suction compressors, ducting, control valves, and exhaust ducting.

WEIGHTS

General and weight data for the D3.2-3A baseline concept are presented in Table A-2. The configuration was sized to a takeoff gross weight of 769,000 pounds to achieve a range of 6,500 nautical miles.

TABLE A-2
D3.2-3A CONCEPT GENERAL AND WEIGHT DATA

GENERAL DATA	
MACH NUMBER	3.2
RANGE (N MI)	6,500
FUEL TYPE	TSJF
NUMBER OF PASSENGERS	300
TAKEOFF GROSS WEIGHT (LB)	769,000
MAXIMUM ZERO FUEL WEIGHT (LB)	297,800
MAXIMUM SPACE-LIMITED PAYLOAD (LB)	66,500
WING AREA – TOTAL PLANFORM (FT^2)	9,500
HORIZONTAL TAIL AREA – TOTAL PLANFORM (FT^2)	733
VERTICAL TAIL AREA – TOTAL PLANFORM (FT^2)	670
NUMBER OF ENGINES	4
MAXIMUM SEA LEVEL STATIC DRY THRUST PER ENGINE (LB)	29,500
WEIGHT DATA (LB)	
STRUCTURES	98,054
POWER PLANT	35,115
SYSTEMS	91,066
MANUFACTURER'S EMPTY WEIGHT	224,235
OPERATOR ITEMS	7,100
PAYLOAD	61,500
ZERO FUEL WEIGHT	292,835
FUEL	476,165
TAKEOFF GROSS WEIGHT	769,000

LRC005-B77

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA CR-4375		

4. Title and Subtitle	5. Report Date
1989 High-Speed Civil Transport Studies	May 1991
	6. Performing Organization Code
7. Author(s)	8. Performing Organization Report No.
HSCT Concept Development Group	
	10. Work Unit No. 537-01-02-01
9. Performing Organization Name and Address	11. Contract or Grant No.
McDonnell Douglas Corporation Douglas Aircraft Company 3855 Lakewood Blvd. Long Beach, CA 90846	NAS1-18378
12. Sponsoring Agency Name and Address	13. Type of Report and Period Covered
National Aeronautics and Space Administration Langley Research Center Hampton, VA 23665-5225	Contractor Report
	14. Sponsoring Agency Code

15. Supplementary Notes

NASA Langley Contracting Officer Technical Representative: Donald L. Maiden

16. Abstract

This report contains the results of the Douglas Aircraft Company system studies related to high-speed civil transports (HSCTs). The tasks were performed under a 1-year extension of NASA Langley Research Center Contract NAS1-18378.

The system studies were conducted to assess the environmental compatibility of a high-speed civil transport at a design Mach number 3.2 Sonic boom minimization, exterior noise, and engine emissions have been assessed together with the effect of a laminar flow control (LFC) technology on vehicle gross weight.

The general results indicated that (1) achievement of a 90-PLdB sonic boom loudness level goal at Mach 3.2 may not be practical, (2) the high-flow engine cycle concept shows promise of achieving the side-line FAR Part 36 noise limit but may not achieve the aircraft range design goal of 6,500 nautical miles, (3) the rich-burn/quick-quench (RB/QQ) combustor concept shows promise for achieving low $EINO_x$ levels when combined with a premixed pilot stage/advanced-technology high-power stage duct burner in the P&W variable-stream-control-engine (VSCE), and (4) full chord wing LFC has significant performance and economic advantages relative to the turbulant wing baseline.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
High-Speed Civil Transport Sonic Boom Exterior Noise Engine Emissions LFC	Unclassified - Unlimited Subject Category 05

19. Security Classification (of this report)	20. Security Classification (of this page)	21. No. of pages	22. Price
Unclassified	Unclassified	92	A05

* For sale by the National Technical Information Service, Springfield, Virginia 22161